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SPACE CENTER

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ANCHORED INTERPLANETARY MONITORING PLATFORM
(AIMP)
SPACECRAFT HANDLING PLAN

Prepared by
Delta Spacecraft Operations Office - ULO

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A. MISSION OBJECTIVES

The primary objectives of the Anchored Interplanetary Monitoring Platform (AIMP) project are to investigate, in the vicinity of the moon, the characteristics of the planetary magnetic field, solar-plasma flux, and solar and galactic cosmic rays, as well as to study the magnetohydrodynamic wake of the earth in the interplanetary medium at lunar distances every 29.5 days.

Secondary objectives of the AIMP project, accomplished by analyzing respectively the dynamics of the spacecraft orbit and the variations of its telemetry signal, are to provide information on the lunar gravitational field (for investigating the mass distribution of the moon, the earth-moon mass ratio, and the figure of the moon), and on any possible lunar ionosphere by observing radio wave propagation from the vicinity of the moon.

The lunar anchored satellite is an improved means of achieving program objectives since it traverses the sunlit and night sides of the earth once every 29.5 days.

Studies indicate that a lunar orbit inclined between 140 and 180 degrees to the moon's equator, having an apocynthion less than 46,000 km and a pericynthion greater than 500 km, is acceptable for probability of success and lifetime criteria; however, a more desirable set of orbital parameters can be obtained through proper selection of retromotor firing time. The actual orbit, within the ranges selected, will be determined by the retromotor firing time which will be determined from an analysis of the actual transfer trajectory attained. The AIMP Project Office will use real time trajectory tracking data to select the firing time with the best set of orbital parameters for meeting project constraints (life, spin-axis/sun-angle, and shadow conditions) and scientific objectives.

The AIMP spacecraft (IMP-D) will be launched from the Eastern Test Range (ETR) during the second quarter of 1966 into a transfer trajectory by a thrust-augmented improved Delta (DSV-3E) with an FW-4D solid propellant third stage.

B. SPACECRAFT DESCRIPTION

The main structure of the AIMP spacecraft is a two-piece magnesium axial-thrust tube, with a Delta attach flange on one end and a retromotor flange on the other. Surrounding this main supporting structure is an octagonal aluminum-honeycomb equipment deck with eight radial support struts, an aluminum-honeycomb top cover, four support brackets and arms for solar cell paddles, and two support brackets and booms for fluxgate magnetometers. The octagonal equipment deck is 27 inches across the flats and 8.6 inches high. The retromotor extends 21.6 inches above the equipment deck. Figure 1 is a side view giving overall dimensions of the spacecraft. Figure 2 is a top view of the spacecraft showing the solar paddles and fluxgate booms extended to their maximum lengths. Experiments and electronics are mounted on the periphery of the equipment deck in modular support frames. The top cover encloses all equipment mounted on the deck, provides an RF ground plane, and furnishes exterior surfaces for the passive thermal-control coatings. Four single RF pivoting turnstile antennas screw into cups protruding through the cover.

The retromotor, bolted to an epoxy-fiberglass adapter ring, is cantilevered from the top flange of the thrust tube. The motor case is mounted by a bolt-lug arrangement that accommodates motor case thermal expansion and provides adjustment for motor/spin-axis alignment. The fluxgate booms project from the lower surface of the octagonal equipment deck. All structural components are made from materials of low permeability (less than 1.001) to meet the magnetic cleanliness requirements of the fluxgate experiments.

In the launch configuration, the solar paddles and fluxgate booms are folded and secured along the FW-4D third stage. The folded fluxgate booms are located within the area formed by the folded solar paddles. Figure 3 is a dimensional view of the spacecraft and retromotor, with the solar panels and magnetometer booms folded.

The solar paddles and fluxgate booms are deployed after third stage burnout but before third stage separation (approximately 1 minute after the spacecraft is injected into the transfer trajectory). After third stage separation, this configuration will satisfy the requirement that the ratio of the roll moment-of-inertia to the pitch moment-of-inertia (with the loaded retromotor) will be at least 1.20 during the long coast.

The total spacecraft weight is approximately 81.5 pounds.

C. SPACECRAFT SUBSYSTEMS

1. Stabilization. The launch vehicle guidance system provides stabilization through first and second stage burn and the coast phase pitch program. At the end of the coast phase pitch program, spin rockets located on a spin table connection between the second and third stage will spin the spacecraft/third stage package to a nominal 150 rpm. The expended second stage motor is jettisoned after spinup. After the third stage burnout, a yo-yo despin mechanism will reduce the spin rate to approximately 100 rpm. Erection of the solar paddles and the fluxgate booms will further reduce the spin rate to 27 ± 2 rpm. After spacecraft/third stage separation, the spacecraft retromotor combination will have a roll/pitch moment-of-inertia ratio compatible with the spin rate for adequate spin stabilization for the transfer trajectory coast period. A rocket tumble system will tumble the FW-4D to reduce the possibility of spacecraft/vehicle collision after third stage separation.

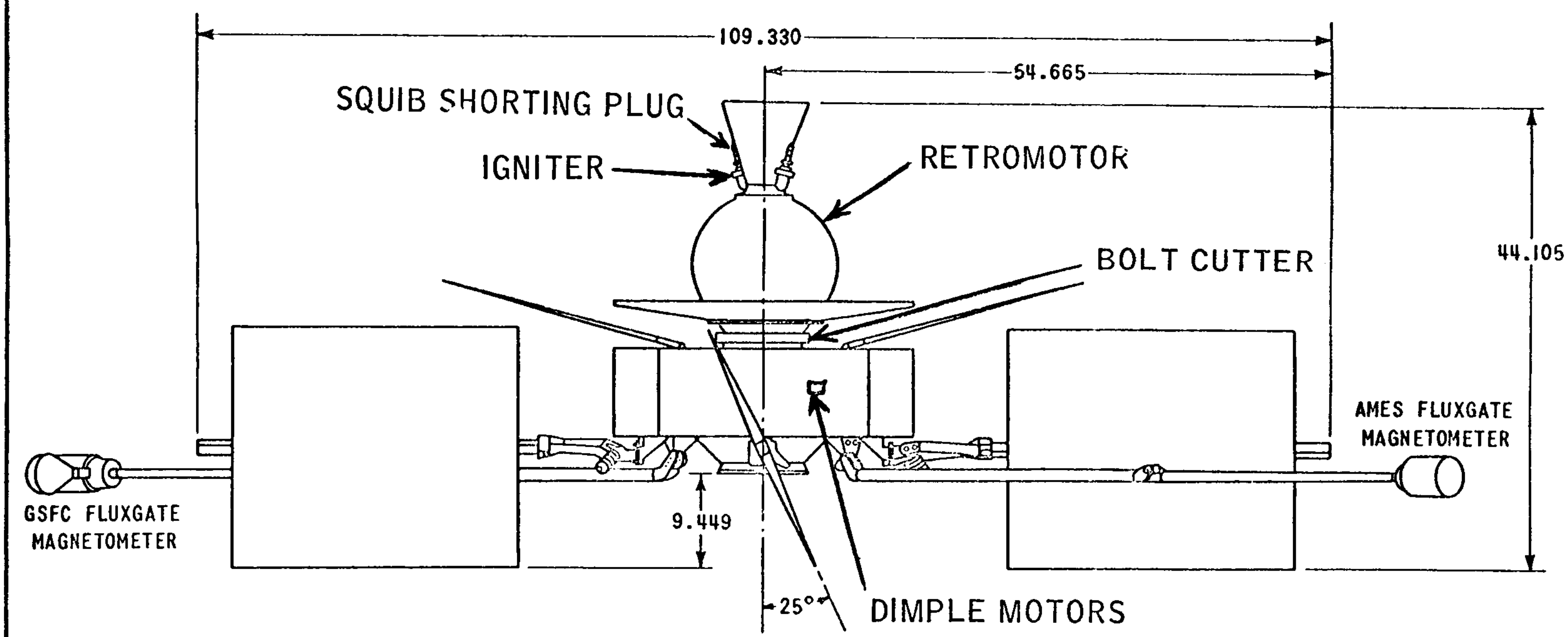


Figure 1. AIMP Spacecraft, Side View

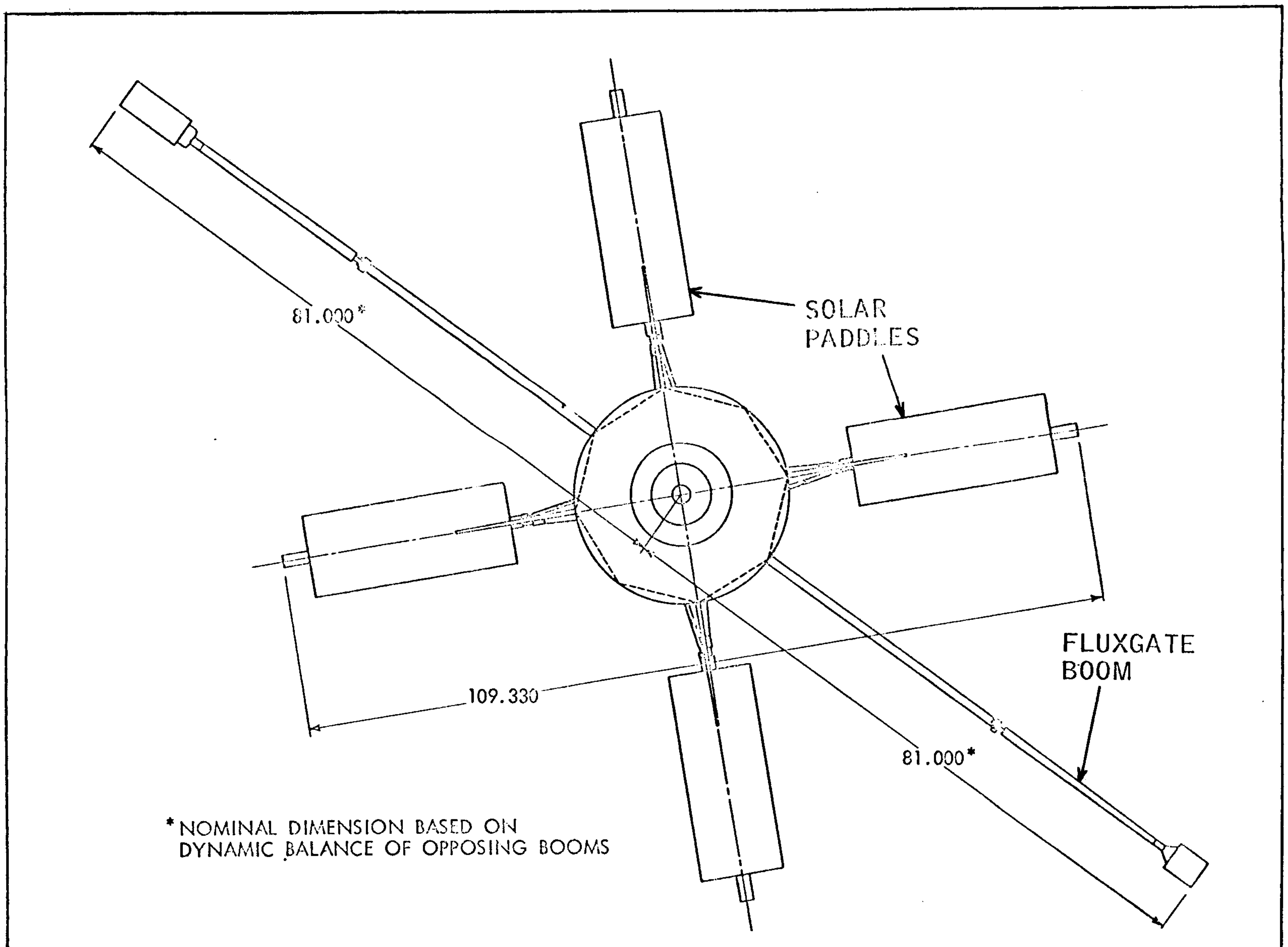


Figure 2. AIMP Spacecraft, Top View

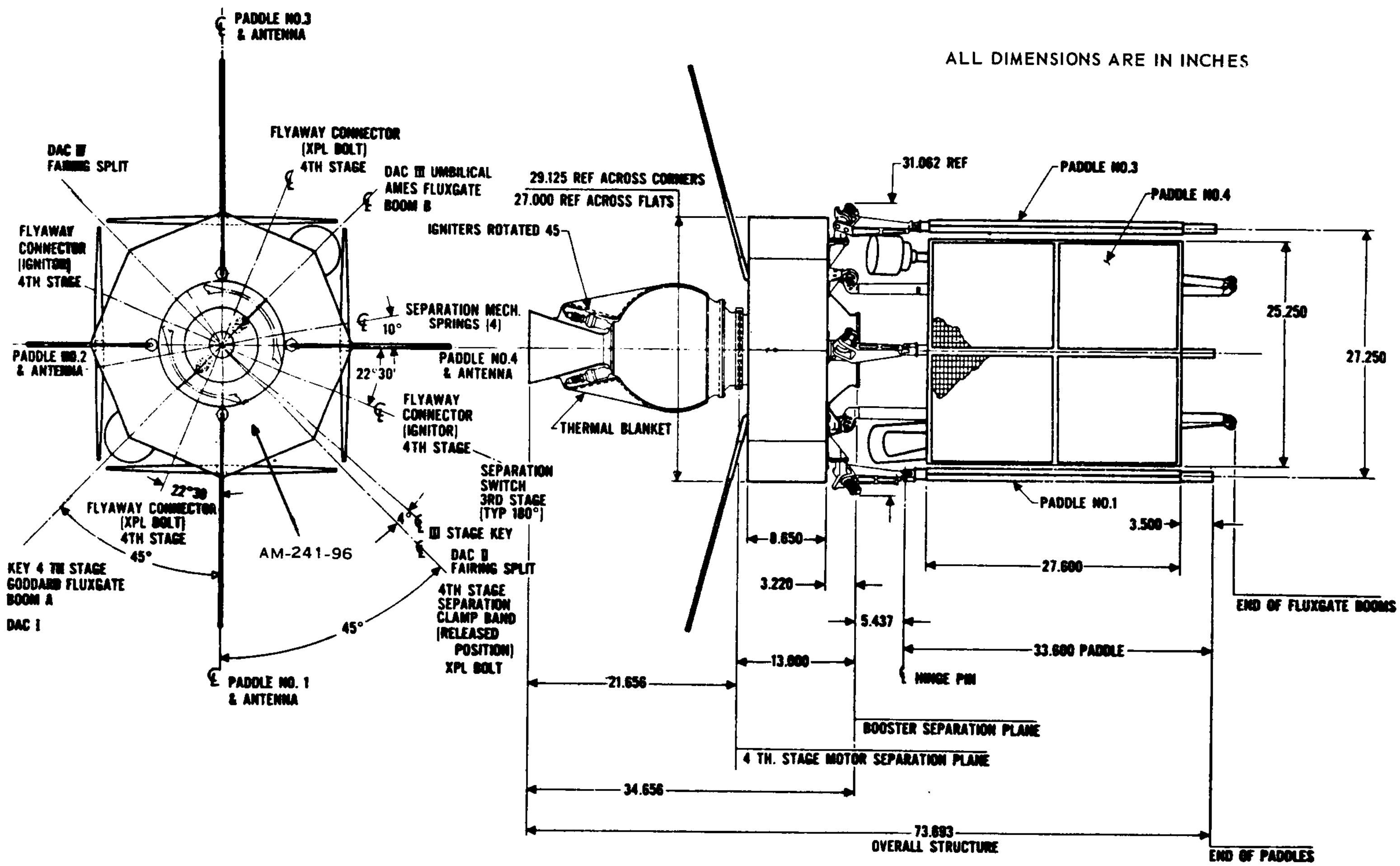


Figure 3. AIMP Outline and Dimensional Drawing

At the end of the transfer period, the retromotor will be fired either by direct command or by initiating the on-board retromotor timer by ground station command. After retromotor burnout and separation, the spacecraft will have an even more favorable moment-of-inertia ratio for maintaining orbital stability.

To meet those requirements, the combined third stage/retromotor/spacecraft will be spin balanced at the Delta spin test facility. The spacecraft and the retromotor will be individually balanced, both statically and dynamically, before assembly with each other and with the third stage motor.

2. Retromotor. The retromotor (figure 4), a Thiokol Chemical Company TE-M-458 solid fuel motor using an ammonium perchlorate polyurethane composite propellant, will be used to slow the spacecraft to a velocity that will permit it to be captured in the lunar gravitational field.

The spherical retromotor case, fabricated from titanium (6 AL-4V), is afforded thermal protection by a vitreous silica-phenolic lining and a high density graphite material throat insert. An asbestos filled polyisoprene rubber insulator containing an integral separation boot is used to relieve thermally induced strain on the propellant grain. The retromotor case, including the igniters and nozzle, will be insulated to maintain allowable temperatures during the transfer trajectory period. The retromotor has a burn time of approximately 20 seconds. Spacecraft/retromotor separation is accomplished by an explosive bolt device actuated by ground command, or automatically by the retromotor timer 2 hours after ignition.

3. Thermal Control. Thermal control is achieved through a passive system of surface coatings, insulation, and structural conduction-path design which will maintain temperature gradients and excursions within acceptable limits at the selected solar aspect angles and orbital conditions.

Selection of a transfer trajectory launch window which restricts the solar-aspect angle of the spacecraft/retromotor package to an angle between 30 degrees and 150 degrees, combined with the absorptivity/emissivity ratio of the spacecraft, will keep the spacecraft at suitable temperatures.

Two thermal blankets of superinsulation will keep the retromotor propellant and retromotor igniters within allowable limits (between 20°F and 140°F) during the transfer-trajectory phase. The blankets also protect the spacecraft from thermal soak-back after the retromotor has fired. The main blanket is shaped like a truncated cone and is placed over the case and exterior surface of the exit nozzle of the retromotor. The second blanket is circular to fit under the spherical case of the retromotor and is attached to the retromotor adapter. Both thermal blankets are of the same multilayer construction. Ten layers of .0005-inch embossed aluminized Kapton film are alternated with 11 layers of .0015-inch Tissueglass insulation material. These 21 layers are sandwiched between .002-inch sheets of smooth (not embossed) Kapton film.

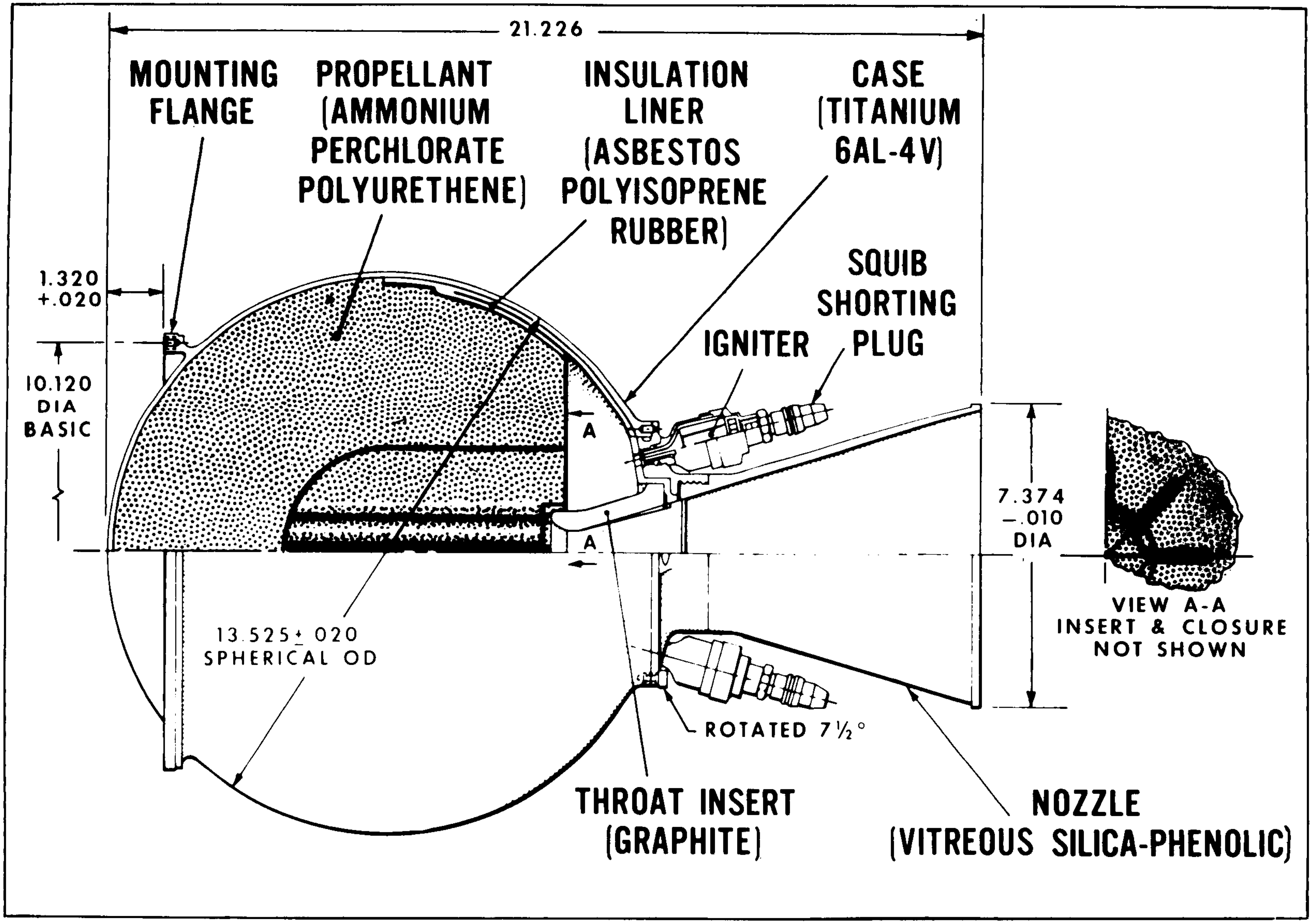


Figure 4. Retromotor, Side Cutaway View

Electrical continuity on the Kapton film is maintained through the vaporized aluminum coating on the outer edges. The layers of Kapton for the main thermal blanket are cut from a pattern which is laid on two overlapping sheets of Kapton film which have been taped together with aluminized Kapton tape. The edge effect of electrical continuity is preserved on each layer of the main blanket. Because the circular blanket is cut from the center of the Kapton sheet, the edge of continuity is removed. To obtain electrical continuity between Kapton layers of the circular blanket, aluminum grounding straps in two places are used to connect both surfaces of each layer of Kapton to the same grounding point. Both blankets are attached to the spacecraft with screws which pass through metal lacing eyelets that are on each blanket. Each surface of every layer of the aluminized Kapton on both blankets is grounded to a minimum of two common points, which are then grounded to the gantry or spacecraft. For the main blanket, the aluminized edge provides electrical continuity between the two surfaces of each Kapton layer. To insure grounding of each layer to a common point, a 1 inch square of Tissueglass is removed from every layer at two locations for the lacing eyelets positioned 180° apart. Grounding straps are attached to these two lacing eyelets and then connected to the gantry ground. These eyelets are grounded to the retromotor adapter which is electrically continuous with the spacecraft ground. During the construction of each blanket, the continuity of every aluminized surface is checked with a meter to insure proper grounding.

During the illuminated portion of the lunar orbit (after retromotor separation), maximum component temperature should not exceed 50°C; minimum temperatures should not fall below -15°C. For shadow periods of 1 hour or less (without power lockout), facet temperatures should not fall below -20°C; however, for longer shadow periods (up to 2.7 hours), facet temperatures in the worst case (when the spacecraft enters the shadow at a solar aspect angle of 180 degrees) can drop as low as -35 to -45°C.

4. Telemetry Data System.

a. Telemetry Encoder. The telemetry encoder processes data from the various experiment sensors and spacecraft performance sensors (both analog and digital). The encoder design is a unique compromise between weight, size, and power that provides the required data handling capabilities within the limitations prescribed by spacecraft and launch constraints.

The encloder consists primarily of a voltage controlled oscillator, a 17-level frequency synthesizer, six 32-bit accumulators, and the spacecraft clock. The voltage controlled oscillator converts the analog data and the frequency synthesizer converts the digital data to a frequency. Four of the 32-bit accumulators accommodate the Univeristy of Iowa electron and proton experiment data; the other two are used in the University of California energetic particle flux experiment. A simple parity check of the optical aspect data is performed by counting video pulses. The spacecraft clock and binary frequency division provide the required timing signals and sync pulses. Encoder temperature and analog oscillator calibration is monitored and subcommutated in two words of the system performance parameters. Onboard oscillator calibration is accomplished by use of a standard reference voltage (zener standard reference source) and by use of precision wire-wound resistor voltage dividers to produce four related calibration voltages.

b. Performance Parameters. The spacecraft performance parameters package includes the circuitry necessary to convert the outputs of the various spacecraft temperature, voltage, and current sensors to inputs acceptable to the analog-to-digital converter of the encoder. The performance parameters package also contains the 7-volt power source for the temperature transducers, and controls the dumping point voltage of the solar array regulator.

5. Telemetry Communications System. The spacecraft communications system provides radio links for receiving range and range-rate and ground command signals, and for transmitting telemetry or tracking data from the spacecraft. The system consists of a telemetry transmitter, a range and range-rate transponder, a redundant command receiver/command decoder unit, and an antenna and hybrid system.

a. Transmitter. The transmitter functions as a 136.020-mc, 6-watt, pulsed frequency modulation/pulse modulation (PFM/PM) system with a signal compatible with the STADAN phase lock receiving systems. The transmitter accommodates both the output of the telemetry encoder and the output of the range and range-rate transponder on a time sharing basis.

b. Range and Range-Rate Transponder. The range and range-rate transponder operates in conjunction with the spacecraft transmitter and one of the command receivers to provide ranging data. Either the tone ranging or the hybrid ambiguity resolving ranging system, or both, can be used. The range data information transmitted from the spacecraft will be centered around a sideband 905 kc from the 136.020-mc carrier.

c. Command Receivers and Decoders. The command system includes two command receivers and two command decoders. The command system will use a frequency of 148 mc transmitted by the 10-kw transmission/reception VHF antenna system of the range and range-rate system. The signal will be received by the turnstile telemetry antenna (zero db gain) on the spacecraft and fed to the redundant command receivers which have a sensitivity of -118 dbm.

d. Antenna and Hybrid. The spacecraft antenna system consists of four dipole antennas spaced 90 degrees apart on the upper surface of the octagonal cover and elevated at an angle of 15 degrees from the attaching point to form a canted turnstile. The antennas are fed in phase quadrature to produce a standard IEEE radiation pattern; i. e., left-hand circularly polarized from the top, linearly polarized from near the center, and right-hand circularly polarized from the bottom. The antennas are fed from a 90-degree stripline coupler/duplexer.

The transmitter and one command receiver are connected to the antenna through the duplexer; the second command receiver through a separate coupler. Transmitter output is fed through the coupler to opposing antenna pairs connected by a half-wavelength cable. The signals to each pair are equal in power but 90 degrees out of phase.

6. Power System.

a. Solar Array. Conversion of solar energy by 7680 2 x 2 cm silicon cells, located on four 25.25-inch wide by 27.6-inch long paddles extending on arms from the equipment deck, provides power to the AIMP. The paddles are oriented to allow a near-uniform solar-cell projection area at any spacecraft solar attitude. The solar cell requirement was established by considering the variation in spacecraft and experiment power demands during light and shadow periods, and by a statistical analysis of probable cell failure. The cells are connected in series-parallel arrangements based on the probable failure mode of the cells in conjunction with the voltage-current requirements of the spacecraft. Table 1 shows solar array power output.

b. Solar Array Regulator. The solar array regulator functions to prevent excessive voltage generated by the solar cells from damaging the spacecraft and its battery. The regulator will operate in either of two stable states depending upon the state of charge of the battery. A current sensor in the battery charge circuit determines the state of charge. When the battery requires charge currents in excess of 100 milliamperes (ma), an appropriate signal to the regulator will set the regulation at 19.6 volts. When the battery charge current diminishes to less than 50 ma, the regulation will be at 18.3 volts, the purpose being to eliminate the possibility of cell unbalance such as would occur at the higher potential.

Table 1. Solar Array Power Output (watts)

	Normal Operation (19.6 v)		Shadow Emergence (15.5 v)	
	Average Minimum (rotation)	Instantaneous Minimum	Average Minimum (rotation)	Instantaneous Minimum
Initial	66.1	54.4	57.9	47.6
End of life	56.2	46.3	49.2	40.5

c. Battery. The storage battery is a sealed nonmagnetic silver cadmium battery, rated at 11 ampere-hours. It is composed of 13 cells in series, which provides a battery discharge voltage, when fully charged, of 15 volts (1.15 volts/cell) and 12 volts (0.93 volts/cell) when fully discharged. Discharge below the 12 volts is detrimental to the battery and accordingly an undervoltage relay is used to disconnect the loads from the system below this value. The maximum safe charge is 19.6 volts. This allowable safe limit is maintained by a battery charge regulator which operates by shunting the solar array output. Continuous charging at 19.6 volts after the battery is fully charged (battery accepts less than 100 ma of current) is considered detrimental to the battery and therefore the regulator is designed to reduce the charge voltage to 18.3 volts upon full charge of the battery, thus reducing charge current to a trickle rate.

d. Prime Converter. All major experiment and instrument power will be supplied from the prime converter; only control systems such as the undervoltage and solar array regulator will operate directly from solar array or battery power. The prime converter will provide three regulated voltages under the load conditions specified in table 2.

Table 2. Prime Converter Characteristics

Voltage	Regulation (1%)	Load (Milliamperes)			Minimum Short Circuit Capability
		Min.	Avg.	Peak	
12	±1	600	620	650	200% (of largest fuse rating)
20	±1	80	100	260	200% (of largest fuse rating)
28	±1	590	590	590	None

7. <u>Programmers</u>.

a. Undervoltage Detector. The undervoltage detector consists of a voltage sensor, 4-hour timer, and appropriate control and switching circuitry. The detector monitors the battery voltage and turns off the prime converter power when the battery voltage drops below 12 ± .25 volts. When turned off, the prime converter power remains off until turned on by the 4-hour timer or by the battery voltage returning to 18 volts. If, at the end of the 4-hour period, the battery voltage is not greater than 12 ± .25 volts, the undervoltage detector will reset the timer for another 4-hour period. To permit battery charging for periods of 4 hours, regardless of system voltage level, the 18-volt turn-on function is disabled when the undervoltage detector is actuated by ground command.

b. Retromotor Timer. The retromotor electronics package is a redundant set of 2-hour timers initiating retromotor ignition and retromotor separation (both timers perform both functions), and a direct command ignition and separation circuit. The timers operate on ground command at the appropriate time during the transfer trajectory period.

c. Flipper Control and Retromotor Burn Timer. The flipper control applies power to each individual magnetometer flipper (sensor rotating device) alternately for a period of 10 minutes every 12 hours. Power lines between the flipper control and each experiment flipper are individually fused to prevent failure of one flipper from interfering with the operation of the other. Control time determination is obtained from the encoder timing pulses. Flipper operating power comes directly from the solar array battery combination.

The retromotor performance monitor is a thrust actuated switch operated by the deceleration caused by retromotor ignition, plus a timing circuit that monitors retromotor burn time. This information, transmitted to the ground as part of the performance parameter telemetry, will be used with tracking data for calculating total retromotor impulse.

d. Despin Function. The DAC third stage sequence timer package will provide the power pulse to the yo-yo despin dimple motors on the spacecraft through a flyaway connector in the third stage interface.

8. Optical Aspect System. The optical aspect system consists of four sensors and associated electronics. A 180-degree field of view digital output solar aspect sensor will be used to determine the angle between the spacecraft's spin axis and the sun. The sensor is capable of resolving the spin-axis/sun-angle within ± 0.5 degree.

The remaining three sensors are pencil-beam (0.25 degree field-of-view) telescopes mounted at angles of 45 degrees, 90 degrees, and 135 degrees to the spin axis. The sensors, if actuated by reflected light from earth or moon, provide digital outputs reflecting the angle subtended by the moon or the earth from each sensor angle and the angle through which the spacecraft rotates between sun sensing and earth or moon horizon sensing (centerpoint of the three sensors). In addition, one of the sensors will provide a digitalized video signal of approximately a 10-degree portion of its sweep across the moon's (or earth's) surface.

The optical aspect (OA) system will furnish a sun pulse or, in shadow, a pseudo sun pulse, to the experiments requiring it.

If the transfer-trajectory injection point occurs when the sunlit earth is observable by the optical aspect telescopes, the telemetered data will help to determine the spacecraft's actual orientation in space. This unambiguous information will also be used as one of the inputs to calculate retromotor firing time. If injection occurs at a time when the earth is in shadow as seen by the OA detectors, the predicted spin-axis/sun-angle will be compared with sensor spin-axis/sun-angle data to determine the approximate spacecraft orientation.

When the spacecraft is in lunar orbit, sensing the edge of the moon and the associated low resolution video scan of its surface may make it possible to compute spacecraft spin axis orientation very accurately.

D. SPACECRAFT EXPERIMENTS

1. Radiation Experiments. The AIMP complement of experiments includes three monitoring experiments:

1 The energetic particle flux experiments conducted by the University of California-K. A. Anderson, investigator

2 The electron and proton experiment conducted by the University of Iowa-J. A. Van Allen, investigator

3 The plasma probe conducted by the Massachusetts Institute of Technology-H. S. Bridge, investigator

a. Energetic Particle Flux Experiments (University of California). Objectives of these experiments are as follows:

1 Search for low energy solar electrons in interplanetary space. Fluxes as small as about 3 cm^{-2} sr^{-1} sec^{-1} will be detectable. The main interest here is with electron fluxes inside gas clouds which produce magnetic storms. If this aspect of the experiment is negative, at least a very small lower limit will have been set.

2 Low energy protons can be detected and the relation of these to storm producing solar plasma will complement the electron measurement.

3 The flux of terrestrial electrons and protons sluffed off the magnetosphere which appear near the moon will be monitored. The frequency of appearance and intensity of energetic electrons in the geomagnetic tail will be monitored 60 earth radii (R_e) from the earth.

4 The ion chamber will provide a precise and sensitive monitor of changes in the galactic cosmic ray intensity of all time scales down to 6 minutes or so.

5 The counters will provide a time history of solar cosmic ray events over three integral energy regions:

GM 1	50 kev
Ion chamber	17 Mev
GM 2	0.5 Mev

Spectral changes will appear as changes in the ratios of the three integral rates.

Of considerable practical interest in connection with solar cosmic ray events observed on the lunar anchored IMP satellites will be the properties of the moon as a particle shield. Whether or not particles are excluded from the anti-solar surface of the moon can be answered for the above three energy intervals, each as a function of time. Of further practical interest is that the ion chamber inherently converts the particle spectrum to a dose rate measurement. It is planned that the shielding will be about 0.4 g cm^{-2}.

The monitor apparatus (ion chamber and two GM tubes) will be essentially the same as that flown on IMP-B and IMP-C.

b. Electron and Proton Experiment (Univeristy of Iowa). The objectives of this experiment are:

1 To study the spatial, temporal, and angular distribution of electrons with energies exceeding 40 kev in the magnetospheric wake of the earth at 60 R_e.

2 To search for electrons with energies exceeding 40 kev in the wake of the moon, and to conduct a detailed study of their distribution if workable intensities are found.

3 To study the incidence and intensity of low energy solar cosmic rays versus time profile (protons and alpha particles separately) in interplanetary space, and to determine their energy spectra and angular distribution.

4 To study solar X-rays in the 0-14 angstrom range.

To accomplish these objectives, three GM tubes and one PNJ are used. The GM tubes are eon type 6213 with 1.2 mg/cm^2 mica windows. All three have the same characteristics except for view angle. The collimator for GM 1 is fan-shaped with a 12-degree full angle in the meridian plane and a 30-degree full angle in the equatorial plane of the spacecraft. The collimator of GM 2, a circular cone with a 25-degree half-angle, has its axis parallel to the spin axis. The collimator for GM 3, also a circular cone with a 25-degree half-angle, has its axis parallel to the spacecraft spin axis but directed opposite to that of GM 2. The PNJ, a totally depleted surface barrier silicon detector about 25 microns thick, has its axis in the equatorial plane of the spacecraft and passes through the rotational axis. Its view angle takes the form of a circular cone with a 30-degree angle.

c. Plasma Probe (Massachusetts Institute of Technology). The experiment is designed to measure the following:

1 Angular distribution of the total proton flux in the equatorial meridian plane of the spacecraft.

2 Energy distribution of the proton flux at or near the same angle as the peak of the total proton flux.

3 Angular distribution of the electron flux in the equatorial and meridian plane of the spacecraft.

4 Energy distribution of the electron flux at or near the same angle as the peak of the total electron flux.

The sensor is a current collector of the MIT Faraday cup variety mounted so that its direction of view is at right angles to the spin axis of the vehicle. The sensor has an acceptance cone of 68 degrees. As the sensor rotates with the spacecraft, the variation of the signal with time is determined by the directional characteristics of the plasma in the equatorial plane and by the angular acceptance function of the sensor itself. In particular, the sensor is designed to measure the energy spectrum and angular distribution of the proton and electron flux of the plasma at specific increments in the range of 100 ev to 5 kev.

2. Ames Magnetometer. The Ames magnetometer experiment consists of a boom mounted sensor unit located approximately 7 feet from the center of the spacecraft perpendicular to the spin axis, and the magnetometer electronics located in the equipment area.

The sensor unit consists of three orthogonally mounted individual flux gate sensors, x, y, and z. Sensors x and y are mounted perpendicular to the spin axis; sensor z parallel to the spin axis. A flipper system rotates sensors x and z 90 degrees once each day to re-orient sensor position for calibration.

The system is designed to measure the spatial and temporal variations of the interplanetary and lunar magnetic fields in 3 linear ranges covering the 0.2- to 200-gamma range.

The on-board data processing system uses a synchronous demodulation process to maintain the spacecraft coordinate system, and data compression techniques to provide a more constant output accuracy over the dynamic range of the system. The output of the data processing system is presented in analog form to the spacecraft encoder.

3. GSFC Magnetometer. The GSFC magnetometer experiment is also a boom mounted three component fluxgate magnetometer provided with a flipper mechanism.

The orthogonally mounted sensors are arranged with one sensor parallel to the spin axis of the spacecraft and the remaining two perpendicular to the spin axis. A flipper reorients the sensor array once every 24 hours by rotating one of the perpendicular sensors and the parallel mounted sensor 90 degrees around the axis of the remaining sensor. This achieves the same effect, for calibration purposes, as flipping the spin axis sensor 180 degrees.

The device has a dynamic range of 64-gammas and an individual sensor sensitivity and quantization error (when used with a specially designed 8-bit analog-to-digital conversion unit) of ± 1/4 gamma.

Besides measuring the interplanetary and lunar magnetic fields, this experiment will provide information on the interaction of the solar plasma stream with any lunar magnetic field.

4. Thermal Ion Experiment (GSFC). This experiment measures the concentration and temperature of thermal electrons (energies less than a few electron volts) and the concentration, masses, and temperatures of thermal ions over the entire orbit.

5. Solar Cell Damage Experiment. The solar cell damage experiment is a GSFC engineering experiment which will provide information on radiation damage to solar cells and solar cell cover glass of varying thicknesses and compositions, as well as on the protection afforded solar cells by each type of cover glass.

To accomplish this, one facet of the spacecraft will carry a panel of four groups of sixteen 1-cm x 2-cm N-on-P silicon solar cells with a nominal base resistivity of 10 ohms/cm. Mounting the panel on one of the facets permits cell exposure that will provide meaningful data on most spin-axis/sun-line orientations. One group of cells will be unshielded; the second will have an integral 25-micron cover glass, the third, a 6-mil fused silica cover glass; and the fourth, a 6-mil microsheet cover.

The sixteen solar cells composing a group are series connected to a 60-ohm precision resistor of the size required to produce a 4.0- to 4.5-volt output under space conditions with normal illumination. These solar cells are typical production cells of a type which have undergone extensive laboratory testing to determine electron and proton effects. Therefore, output variations among solar cell groups can be related to solar cell or solar cell cover glass damage. A thermistor imbedded in the solar cell panel just under the solar cells will monitor variations in experiment temperature.

To assure experiment accuracy, the cell groups will be calibrated for illumination, angle of incidence, and temperature effect, both after environmental testing and before flight.

6. Passive Experiments. The passive experiments will use the telemetry and the range and range-rate signals as data sources to study selenodesy and propagation of lunar ionosphere radio waves.

A. M. Peterson (Stanford University) will study the spacecraft telemetry signal to determine the effects of the lunar ionosphere on radio wave propagation.

W. M. Kaula (Univeristy of California, Los Angeles) will analyze variations in the range and range-rate tracking data to obtain selenodetic information.

SECTION II
ORDNANCE

A. RETROMOTOR AND IGNITER

1. Retromotor Description. The retromotor (solid propellant) is a spherical design utilizing a modified TE-345 (Titan Vernier) and TE-375 (Syncom) type case and attach ring, a TE-385 (Gemini) and TE-444 (Lincoln Lab) type aft closure and nozzle assembly, and twin TE-P-462, pyrogen igniters.

The retromotor is manufactured by the Thiokol Chemical Corporation of Elkton, Maryland. It is officially designated the TE-M-458 Retromotor and Igniter.

The motor diameter is 13.562 inches and the length is 21.226 inches overall. The chamber is fabricated from 6AL-4V Titanium. The nozzle is a high pressure molding of a viteous silica-phenolic material. A high density graphite material (Graph-i-tite GX) is used as the throat material. The insert is pressed into the aft closure and bonded in place using EPON VIII. For internal insulation of the rocket motor, Gen-Guard V44, an asbestos-filled polyisoprene rubber, is used. The insulator contains an integral separation boot to relieve thermally induced strains in the propellant grain. The total weight of the retromotor is approximately 81.5 pounds. Physical and ballistic characteristics are included in table 3.

2. Igniter Description. The motor is ignited by redundant pyrogen igniters. The TE-P-462 igniter consists of a case and head cap made from 302 stainless steel; a cartridge loaded TP-E-8035 propellant grain; a Holex 4497 one amp, 1 watt, single bridgewire squib; a boron pellet booster charge; and a silica-phenolic nozzle throat section. The igniter is approximately 4.5 inches long, weighs 0.41 pounds, and burns for approximately 0.2 second. Figure 5 is a cutaway view of the igniter assembly. The Holex 4497 squibs have been satisfactorily subjected to the ETR 1 watt or 1 amp for 5 minutes test and the two part electrostatic (ZAP) test consisting of a discharge of a 500 micromicrofarad capacitor at 500 volts and application of 30,000 volts dc across the pin to case mode. See table 3 for further details. Figure 6 is a detailed drawing of the igniter squib.

The igniter assemblies will be shipped separate from the retromotor. Both igniters and igniter squibs in each motor are interchangeable. Flight igniters will be installed in the retromotor just prior to fairing installation. Refer to Appendix A for igniter installation and leak test procedures.

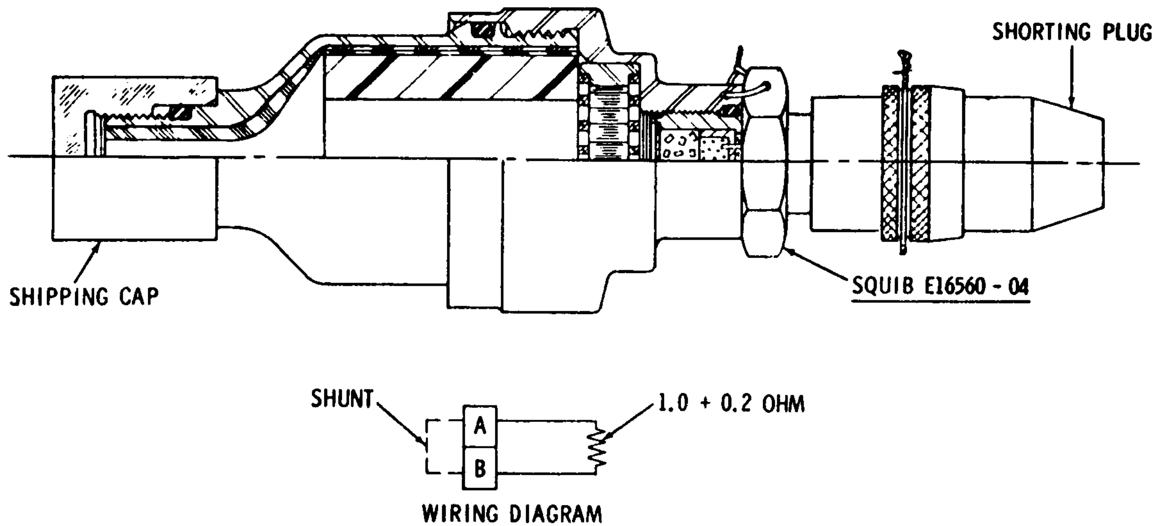


Figure 5. Igniter Assembly

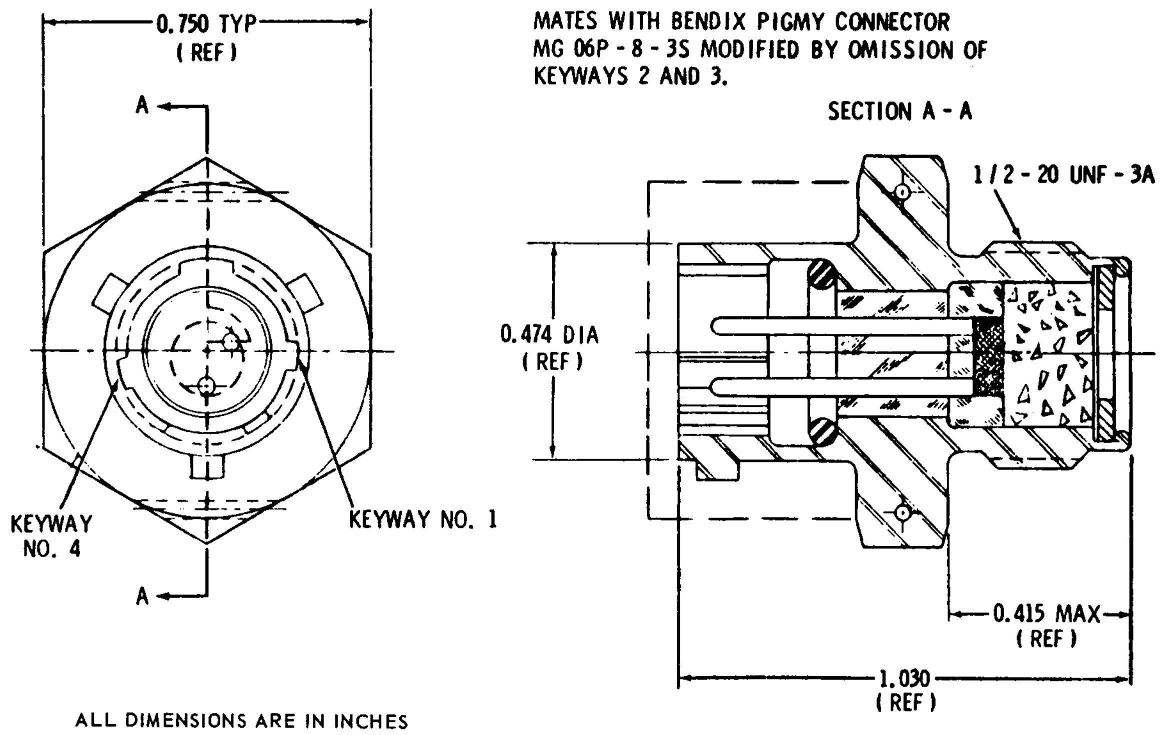


Figure 6. Igniter Squib, Cross-sectional View

Table 3. Retromotor Data

MECHANICAL DATA			
Total Weight - Loaded	81.5 pounds (approx.)		
Inert Weight - Fired	9.6 pounds (approx.)		
Center of Gravity			
Inert Parts - Fired	8.97 inches forward of attach flange		
Loaded Assembly	5.35 inches forward of attach flange		
Propellant Composition	Ammonium perchlorate and aluminum powder with a polyurethane binder		
MOTOR PERFORMANCE DATA			
Pressure and Time Parameters (Nominal)			
Motor Conditioned Firing Temp.	0°F	$+60^\circ$F	$+120^\circ$F
Max Chamber Pressure, PSIA	514	550	585
Average Chamber Pressure, PSIA	468	500	535
Burn Time, sec	23.2	21.8	20.4
Action Time, sec	24.0	22.4	21.0
Ignition Delay Time, sec	0.070	0.060	0.055
Thrust and Impulse Parameters (Nominal)			
Motor Conditioned Firing Temp.	0°F	$+60^\circ$F	$+120^\circ$F
Maximum Thrust, lbf	855	916	976
Burntime, Average Thrust, lbf	795	850	910
TEMPERATURE DATA			
Expected Case Temperature	250°F at burnout (22.4 sec) 400°F at 30 sec up to 780°F max at 160 sec		
Storage Temp limits	0° to 120°F		
Recommended Storage Temp	$80^\circ \pm 5^\circ$F		
Estimated Storage Life	3 years at 80°F		
EXPLOSIVE CLASSIFICATION	ETR Category "A"		

Table 3. Retromotor Data (Cont'd.)

IGNITER	
Igniter Assembly	Thiokol dwg. E17466
Squib (one-igniter assembly)	Thiokol dwg. E16560
Quantity Per Motor	Two igniters 180^{0} apart single bridgewire each.
Squib Specification	Thiokol Spec. SE-225
Bridgewire Resistance	1.0 $\pm$ .2 ohm
Recommended All Fire Current	4.0 $\mp$ .5 amps
No Fire Current	Meets ETR 1 amp or 1 watt for 5-minute no fire requirement.
Insulation Resistance	100 megohms at 500 vdc (pin to pin or pin to case)
Resistance After Firing	1000 ohms minimum (pin to pin or pin to case)
Temperature Limits	-65^{0} to +165^{0}F

3. Firing Circuit Description. TE-M-458 retromotor firing signals are generated in the redundant 1G2 retromotor ignition and separation timer by satisfying a given set of conditions for the spacecraft and the associated ground equipment and following a systematic procedure. The 1G2 timer is controlled by external RF command via the on-board command decoders. Figure 7 is a schematic of the retromotor ordnance circuit.

The following conditions must exist on-board the AIMP Spacecraft in order for it to accept the externally generated RF address tone and three separate execute tones for either direct or 2-hour timed motor ignition:

1 The spacecraft live turn on plug (IS21-P1 & P2) must be inserted. This plug connects the spacecraft batteries to the spacecraft - without it all power is disconnected.

2 The spacecraft ordnance safe plug (IS22-P1 SAFE) must be removed and the ordnance arm plug (IS22-P1 ARM) inserted into the same connector.

3 The ordnance safe test connector (IS22-P2 SAFE) must be removed and replaced by the ordnance arm test connector (IS22-P2 ARM). The short on the igniter remains until the firing relay of the 1G2 timer closes after receipt of appropriate RF command.

4 The spacecraft must be separated from the third stage of the Delta vehicle in order for the 4 switches located on the third stage interface (connected in a series parallel quad) to provide a closed path between the battery to the firing relay to the igniters.

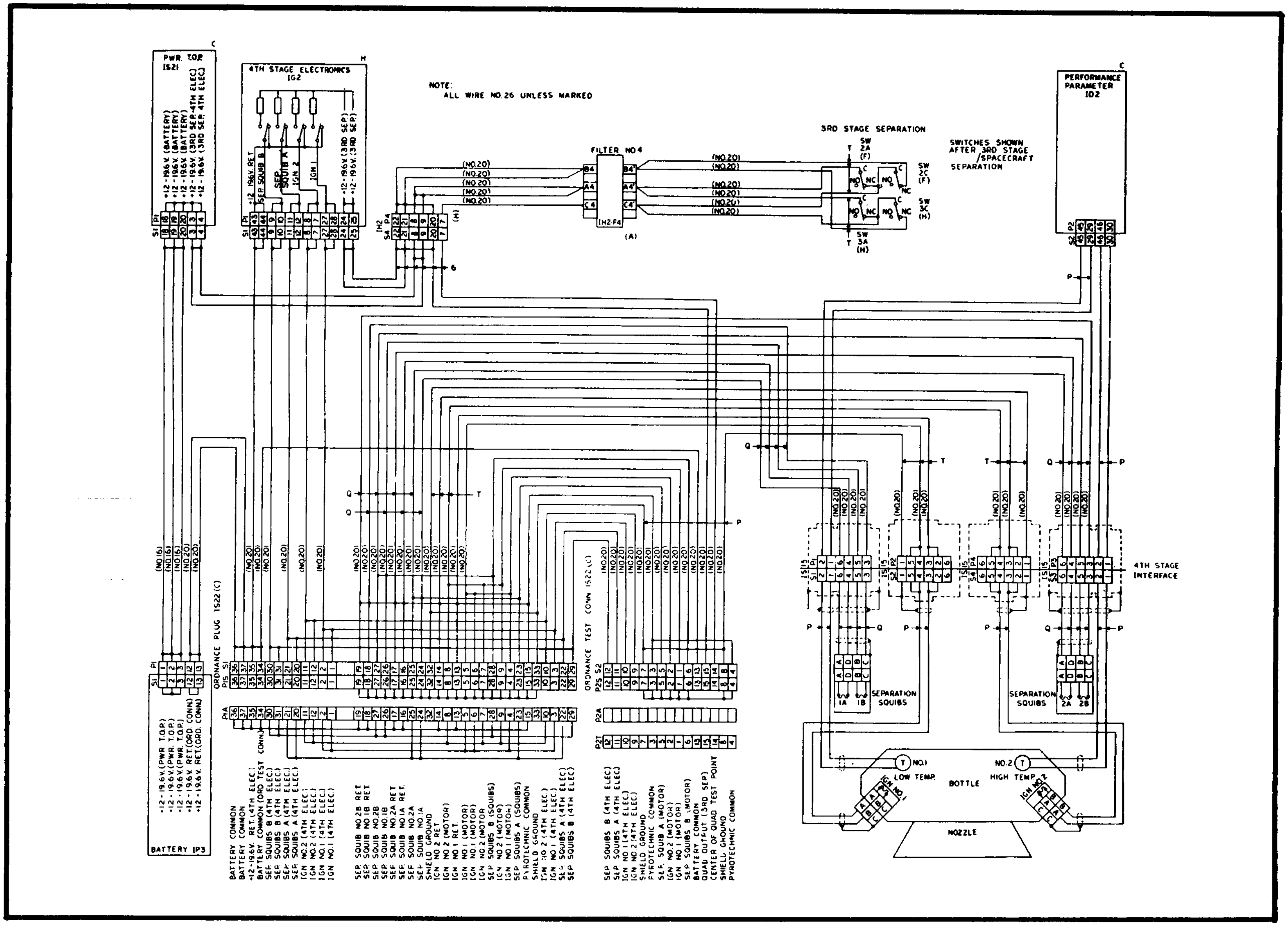


Figure 7. Retromotor Ordnance Circuit

During the motor inspection and assembly operations in the NASA/DAC spin facility, igniters will never be installed in the TE-M-458 retromotor. Neither will they be installed during the spacecraft electrical checks on Gantry 17A up to F-1 day. After the F-1 day spacecraft electrical checks are complete, the two TE-P-462 pyrogen igniters (with shorting plugs) will be assembled into the motor. A brief vacuum leak check will be made through the motor exit cone to insure proper seating of the pyrogen igniter O-rings. The igniter shorting plugs will be removed and the spacecraft igniter harness connected to the igniters. During this operation all RF systems will be off and both ordnance safing plugs will be inserted. The Delta fairing will then be assembled. After the F-0 day spacecraft checkout, the two ordnance arming plugs and the turn on plug will be assembled through the fairing access door.

B. YO-YO WEIGHT RELEASE DIMPLE MOTORS

Four dimple motors (squibs) are located on board the AIMP spacecraft. These dimple motors are used to release two yo-yo weights located 180 degrees apart on the side of the spacecraft. Each release mechanism contains redundant dimple motors. The location of these ordnance items are shown in figure 1. These dimple motors are classified as category "B" ordnance (devices which will not in themselves or by initiating a chain of events cause injury to personnel or damage to property). Each dimple motor case completely confines the products of the explosion so that there is no external flame or flying fragments. Figure 8 is an illustration showing the configuration of a typical dimple motor before and after firing.

Physical and performance characteristics of the dimple motors are given in table 4.

Electrical checks on these dimple motors will be made using an AFETR approved Alinco meter.

Table 4. Dimple Motor Characteristics

PHYSICAL CHARACTERISTICS	
Manufacturer	Hercules Powder Co.
Model No.	DM29A0
Body Size	0.293 inches Dia., 0.51 inch long
Wire leads	#24 AWG, Copper Solid
Case Material	Commercial Brass
Seal	Phenolic
Bridge Resistance	1.4 to 2.6 ohms, wire type
Ignition	Lead Styphnate

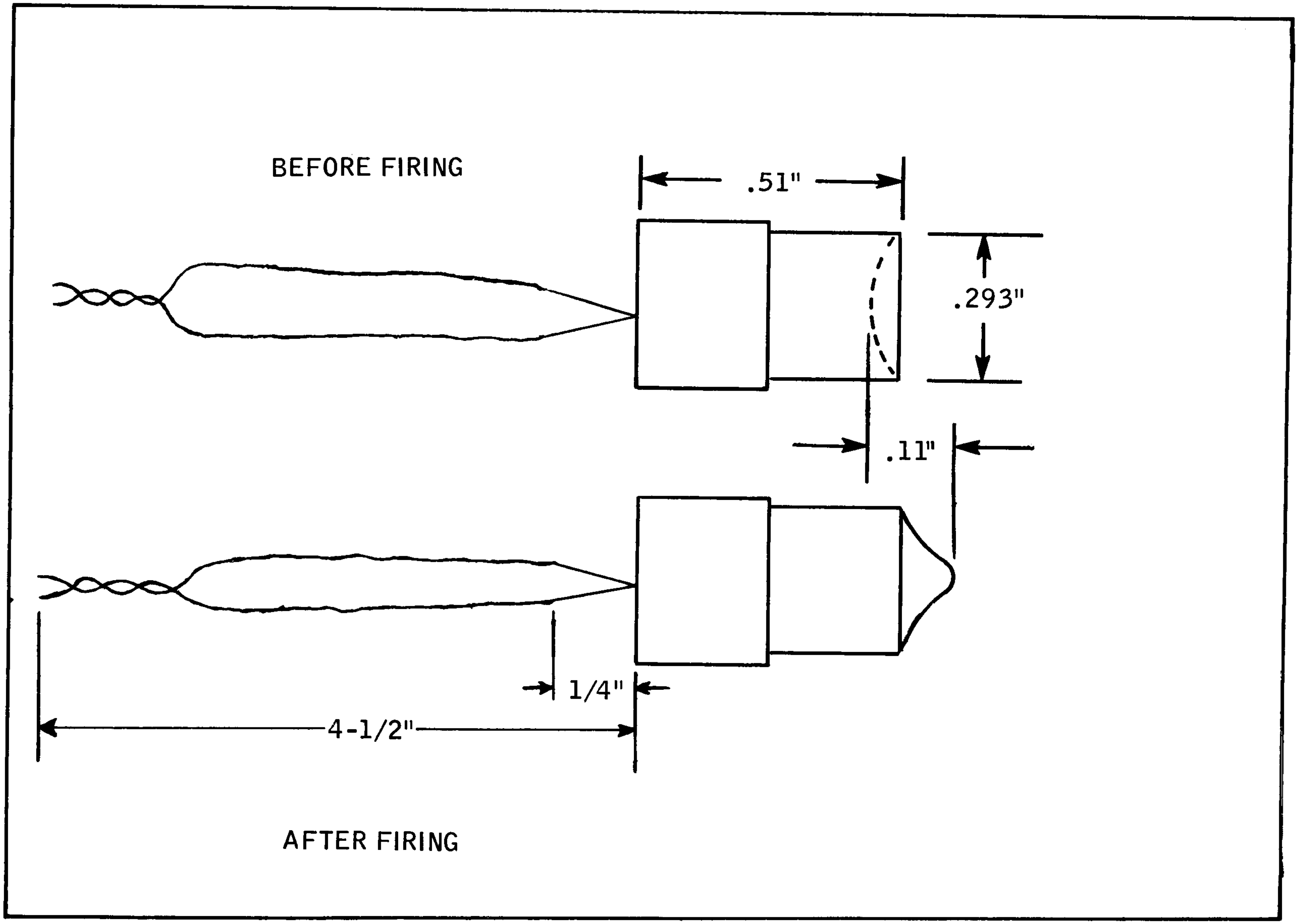


Figure 8. Dimple Motor Configuration Before and After Firing

Table 4. Dimple Motor Characteristics (Cont'd.)

Main Charge	LMNR/Blank powder type
Range Ordnance Classification	Category "B"
Quantity Used Per Spacecraft	4
Location in Spacecraft	Yo-yo weight release mechanisms
PERFORMANCE CHARACTERISTICS	
Max. Nonfire Current	0.25 Amp
Min. Firing Current	0.45 Amp
Recommended Firing Current	1 to 3 Amp
Max. Test Current	10 Milliamps
Ignition Time	Amps 1.0 2.0 3.0 5.0 Ms 4.0 3.5 3.0 2.7
Storage Temperature (High)	Functioned normally after 2 hours at 250°F
Storage Temperature (Low)	Functioned normally at - 65°F
Altitude	Functioned normally at 100,000 feet
Moisture Resistance	Capable of withstanding 100 psi for 24 hours in water
Reliability	99.9 percent
Output	Squib end moves 0.11 inch against an 8 pound spring load in approximately 15 ms.

C. RETROMOTOR SEPARATION BOLT CUTTERS

Two bolt cutters are used to separate the retromotor from the AIMP spacecraft by cutting the bolts in the marman separation clamp. The system is redundant in all aspects. Each cutter has two bridgewires and successful operation of either of the two cutters will accomplish separation. Figure 9 shows a cross sectional view of the bolt cutter. Location of these bolt cutters is shown in figure 1. Physical and performance characteristics of the bolt cutters are given in table 5.

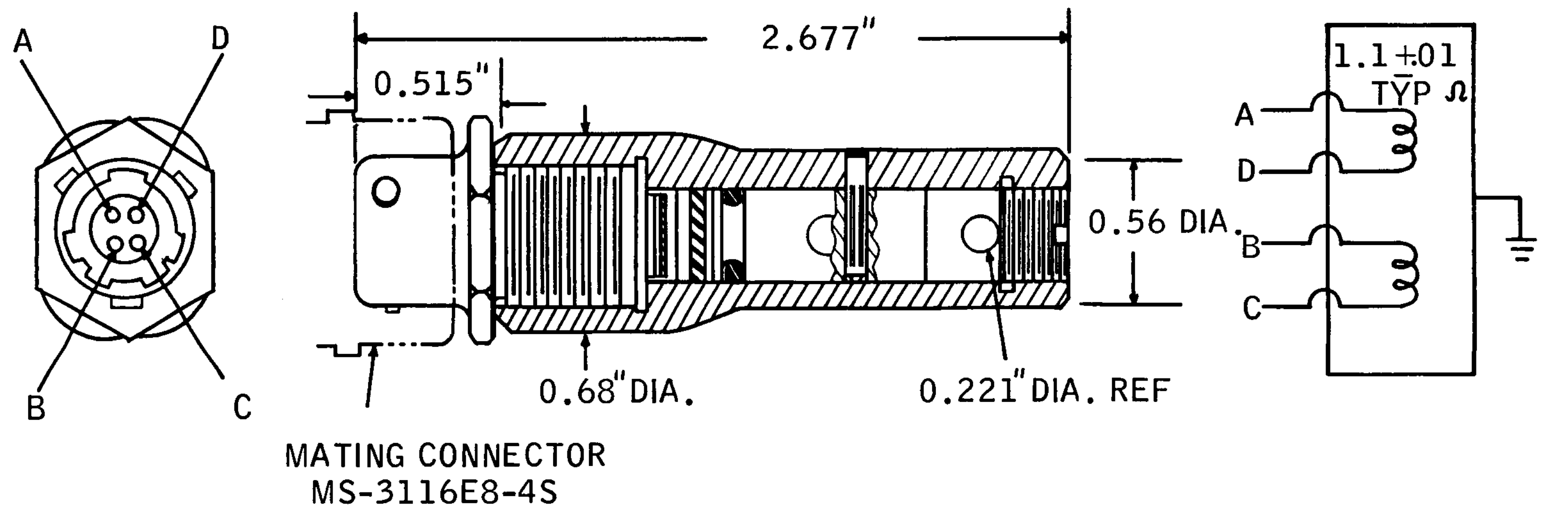


Figure 9. Bolt Cutter, Cross-sectional View

Electrical checks on these bolt cutters will be made using only an AFETR approved Alinco meter.

Table 5. Retromotor Separation Bolt Cutter Characteristics

PHYSICAL CHARACTERISTICS	
Manufacturer	Hi Shear Corp.
Model No.	PC-10
Weight	75 grams (.16 lb)
Quantity Per Spacecraft	2
Capacity	Will cut a .190" dia. bolt of 180,000 psi max tensile strength
Temperature Range	-162° C to +115° C
Humidity	95% @ 30° C
PERFORMANCE CHARACTERISTICS	
Max No Fire Current	1.89 amps @ bridgewire 70° C
Min No Fire Current	3.50 amps @ 70° C bridgewire
Recommended Firing Current	5.0 amps @ 70° C bridgewire
Meets AFETR Ordnance Standard	1 amp, 1 watt
Autoignition Temperature	Approx 610° F
Function time	1 to 3 milliseconds

SECTION III
RADIOACTIVE SOURCES

A. INFLIGHT CALIBRATION SOURCES

The AIMP flight spacecraft will be delivered to the ETR containing one americium-241 (AM-241-96) alpha emitter source (0.01 microcurie) for inflight calibration of scientific instruments (University of Iowa).

The backup spacecraft will be delivered to the ETR containing one americium-241 (AM-241-99) alpha emitter source (0.01 microcurie) (University of Iowa).

The prototype spacecraft will be delivered to the ETR containing one americium-241 (AM-241-72) alpha emitter source (University of Iowa). Table 6 provides additional information on radioactive sources.

B. PREFLIGHT CHECKOUT SOURCES

The following radioactive sources are required for comprehensive preflight checkout of the AIMP spacecraft:

1 Three americium-241 (AM-241-68, -85, and -98) alpha emitter sources, 0.1, 10.0, and 10.0 microcuries respectively (University of Iowa).

2 Eight cobalt-60 (Co-60-97, -101, -102, -103, -107, -108, -109, and -110) beta gamma emitters, 2.1, 0.211, 0.235, 0.230, 1.9, 0.235, 0.235, and 0.224 millicuries, respectively (University of Iowa).

3 One polonium-210 (Po-210-60) alpha gamma emitter, 282.0 millicuries (University of Iowa).

4 Three cobalt-60 (Co-60-91, -92, and -106) beta gamma emitters, 100.0, 100.0, and 500.0 microcuries, respectively (University of California).

C. RADIATION DETECTION INSTRUMENTS AND EMERGENCY EQUIPMENT

1. Radiation Detection Instruments. The following radiation detection instruments will be provided by the GSFC Radiological Office for use at ETR:

1 One Eberline Instrument Corporation, Model E-500B, G-M beta gamma meter. Range: 0-2000 mr/hr.

2 One Eberline Instrument Corporation, Model PAC-1SA, scintillation alpha meter. Range: 0-2x10^6 cpm.

3 One Victoreen Instrument Company, Model 440, low energy survey (beta gamma) meter.

4 One Nuclear Measurements Corporation, Model PC-3A gas flow proportional counter scaler.

2. Emergency Equipment. The following emergency equipment will be provided by the GSFC Radiological office for use at ETR:

1 One W. B. Johnson and Associates, Inc. Model PCS-5, Portable Counting System, providing complete alpha, beta gamma, and neutron detection capabilities, including air sampling.

2 One emergency field kit containing equipment for field area contamination control and decontamination operations.

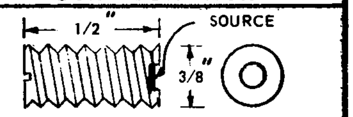
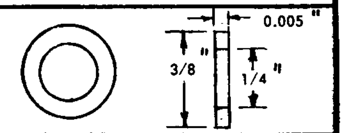
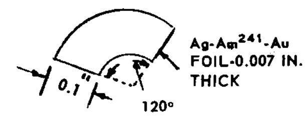
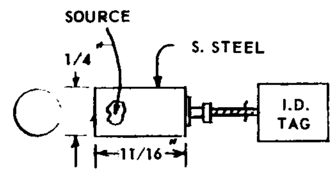
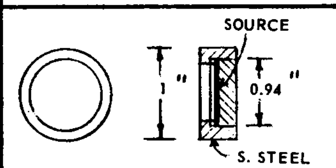
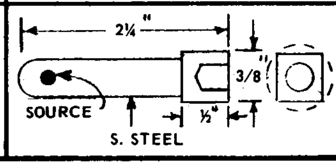
SOURCE MATERIAL	AMOUNT OF ACTIVITY	GSFC SERIAL NO.	EXPERIMENT NO.	MANUFACTURER	MFG MODEL NO.	MFG SERIAL NO.	CONFIGURATION
Americium-241	0.1 uc 10.0 uc 10.0 uc	Am-241-68 Am-241-85 Am-241-98	EI-SN-201 EI-SN-202 EI-SN-203	U. S. Radium	-	Pn-1 #3 #7	
Americium-241	0.1 uc	Am-241-72	EI-SN-201	U. S. Radium	-	-	
Americium-241	0.01 uc 0.01 uc	Am-241-96 Am-241-99	EI-SN-202 EI-SN-203	U. S. Nuclear	-	-	
Cobalt-60	2.1 mc 0.211 mc 0.235 mc 0.230 mc 1.9 mc 0.235 mc 0.235 mc 0.224 mc	Co-60-97 Co-60-101 Co-60-102 Co-60-103 Co-60-107 Co-60-108 Co-60-109 Co-60-110	EI-SN-202 " " " EI-SN-201 " " "	Nuclear-Chicago	-	D15 D57 D17 D32 D6 D33 D39 D45	
Polonium-210	282.0 mc	Po-210-60	U of Iowa	Monsanto Research		MRC-A-55-P-PO 261	
Cobalt-60	100.0 uc 100.0 uc 500.0 uc	Co-60-91 Co-60-92 Co-60-106	U of Calif.	Tracerlab	R-31M		

Table 6. Radioactive Sources

SECTION IV
OPERATIONS PLAN

A. ORDNANCE PREPARATIONS AND ASSEMBLY

The two retromotors and four igniters will be delivered to Cape Kennedy Air Force Station not less than six weeks prior to launch. Upon arrival the motors and igniters are to be placed in storage in the PAA Solid Propellant Storage area until NASA/GSFC and Thiokol personnel are notified and ready to inspect the units.

For any ordnance inspection, preparation, or checkout the appropriate precautions listed under "General Safety and Handling Procedures" (paragraph B of this section) will be followed.

Initially, the motors and igniters will be moved from storage to an area designated by the PAA Solid Propellants facility supervisor for inspection.

The shipping containers will be inspected for any damage incurred during shipment. The motor shipping container will be opened for physical inspection of the motor, nozzle, nozzle post plug, spacecraft attach holes, etc. The initial igniter inspection includes a general physical inspection of the shorting plug, body, exit port, etc.

The igniters will be returned to their respective metal containers and taken to the igniter test area for bridgewire resistance checks. This operation will be performed in the igniter test area where personnel are protected by a steel and concrete barrier. A fit check will be performed using the spacecraft igniter harness (motor fiberglass adapter and harness assembly) and bridgewire resistance measurements made at the fly-away spacecraft connector. The AFETR approved Alinco Tester will be utilized. Upon completion of the bridgewire resistance checks, the shorting plug will be reinstalled on the igniter. Also at the Solid Propellant test facility, a test will be conducted to determine if the retromotor and igniters have detrimental magnetic properties. A Forster/Hoover Magnetometer, type MF-55-331-10X (Appendix D), will be used for this test. If the results of this test indicate the motor and igniters are too magnetic for proper operation of spacecraft experiments, deperming will be necessary. Appendix C is the deperm procedure and Appendix D describes the Forster/Hoover Magnetometer. Appendix E contains a drawing showing the deperm coils around the retromotor/spacecraft combination, and a block diagram depicting the deperm coil/power supply hook-up.

After the motor has been physically inspected, a series of X-rays will be taken and analyzed to insure that there are no separations or cracks in the grain configuration and no discrepancies in the nozzle or case. A dummy (inert) retromotor will be sent to the X-ray facility, for equipment alignment, prior to arrival of the flight retromotor. Thiokol personnel will be available with preshipment X-rays for comparison and consultation. After X-rays have been taken and analyzed, the motor will be placed on the fiberglass motor attach flange and appropriately grounded. The flight multilayered metallized thermal blanket will be fit checked on the retromotor. The retromotor thermal blanket consists of multilayered Kapton with vapor deposited aluminum on both sides. This blanket is slipped over the retromotor and fastened to the mounting flange of the retromotor with 16 stainless steel screws. The blanket, because of aluminum on both sides, is electrically conductive. It also will be continuously grounded during handling, assembly, and after final assembly.

The motors and igniters are not required to follow a critically scheduled sequence but the operations must be completed in a timely manner in order that the motor (without igniters) is available for a fit and alignment check with the AIMP Spacecraft on F-14 days (working) at the NASA/DAC spin balance area.

The retromotor, in its grounded shipping container, will be transported to the NASA/DAC facility by PAA Solid Propellant Personnel approximately 14 working days prior to launch.

After permission has been granted to proceed with operations and the area has been cleared of non-essential personnel, the shipping container cover will be removed and the motor lifted from the case using the air hoist and hydroset. Ground straps will be attached to the air hoist and motor case and will be connected to the building grounding system. Personnel will wear non-static producing smocks and legstats.

The daily schedule of events from F-35 day through F-0 day are listed in paragraph F of this section. Listed below is a sequence of events which must be completed at the spin balance area up through launch.

1. Spin Balance Facility Checks.

<u>1</u> Weigh TE-M-458 retromotor (DAC & GSFC)

<u>2</u> Weigh AIMP spacecraft, booms, and paddles (DAC & GSFC)

<u>3</u> Place retromotor on precision measuring facility (PMF) (GSFC supplied) and measure thrust vector (centroid of throat and exit cone) versus geometric spin axis (GSFC)

<u>4</u> Remove retromotor from PMF and place spacecraft on PMF. Measure two precisely machined surfaces of thrust tube with dial indicator. Measure flatness of fiberglass motor mount (GSFC)

5 Place retromotor on spacecraft on PMF. Align retromotor with dial indicator and calculate total thrust vector versus center of gravity misalignment. Bond two retro flight thermistors to motor case using DC 90-006 primer and aero-space sealant. Allow 12 hour cure (GSFC)

6 Insert dummy (inert loaded) igniters into retromotor. Assemble retromotor thermal blanket using grounding clips. Balnket is to be temporarily sealed at the top to allow later assembly of igniters on F-1 day (GSFC)

7 Move spacecraft/retromotor combination onto third stage. Make a continuity check from DAC attach fitting side through yo-yo despin dimple motors using the AFETR approved Alinco tester to insure proper mating of spacecraft/third stage yo-yo flyaway connector (DAC & GSFC)

8 Assemble four solar paddles and two booms to spacecraft and assemble cord tie-down system. Adjust cord tension to 35 lbs minimum (DAC & GSFC)

9 Dynamically balance spacecraft retromotor/Delta third stage assembly (DAC)

10 Replace conductive asepsis bag over spacecraft (GSFC)

11 Clean inside of DAC transport container with bath of isopropyl alcohol prior to assembly over spacecraft/third stage/spin table assembly (DAC)

12 After container has been placed over spacecraft/third stage assembly, attach dry nitrogen purge system to container and activate. Continuously purge container until unit is placed upon Delta second stage (DAC)

13 Spacecraft electrical turn on and checkout approximately F-5 thru F-1 on Gantry 17A (GSFC)

14 F-2 day remove stripable coating from spacecraft, swab with isopropyl alcohol (GSFC)

2. Safety Features Summary.

1 The igniter squibs will not fire with (a) current of one amp impressed on the squib for a period of 5 minutes or (b) power of one watt impressed on the squib for 5 minutes. The squibs will not fire from the two phase ZAP test.

2 Igniter squibs have a minimum all fire current of 2.32 amps.

3 The pyrogen igniters are not installed until just prior to fairing installation of F-1 day. The igniters are shorted by the shorting plug directly on the igniters at this time.

4 Prior to connecting the harness from the spacecraft to the igniter on F-1 day, resistance and no voltage checks are made to ensure that all safety features are functioning.

5 Redundant mechanical switches interrupt the firing circuit until spacecraft separation from the third stage approximately 20 minutes after liftoff.

6 The timer relay switches short the squibs to ground until receipt of the firing pulse.

7 Two separate non-flight squib shorting plugs (ordnance plugs) remain in place until just prior to sealing the fairing access port.

8 A spacecraft flight turn-on plug is not inserted until just prior to sealing the fairing access port. This plug does not turn the spacecraft on but connects the spacecraft batteries to the spacecraft. Turn on is accomplished just prior to liftoff through the blockhouse umbilical.

9 All firing circuit wiring is twisted and shielded.

10 The connection of the spacecraft harness to the igniters will be made during a no-switching/no-radiation period on F-1 day.

11 During the igniter connection, all non-essential personnel will be cleared from the area. The individual performing the operation will wear nonstatic producing coveralls, legstats, and a wristat and make the connection while standing on a conductive mat connected to the vehicle grounding system.

12 A systematic procedure is required in order to generate a fire signal from the ground checkout equipment. This procedure will be under the direct supervision of the spacecraft Project Manager. All ground command station racks located in the electronic checkout consoles will be physically and electrically removed prior to item 3 above. These individual command stations will be moved to Building AE and remain in a locked area designated by the GSFC Project Manager. As noted in 5 above, if a firing signal was generated after items 7 and 8 have been completed on F-1 day, it is isolated from the igniter squibs until the spacecraft separates from the third stage approximately 20 minutes after liftoff.

13 The design philosophy, operational handling, and safety precautions are similar to those used on the Syncom and Early Bird spacecraft. The feature or capability of inserting the twin pyrogen igniters into the motor just prior to fairing installation on Gantry 17A on F-1 day is an additional safety feature which was not possible on Syncom and Early Bird.

B. GENERAL SAFETY AND HANDLING PROCEDURES.

The following precautions will be exercised during all operations involving the TE-M-458 retromotors and TE-P-462 pyrogen igniters. These instructions are designed primarily to insure maximum safety to personnel but also establish the environmental limits on the motor during storage and insure proper handling of the motor when out of its shipping container.

1. Permission will be obtained from the responsible supervisor of the PAA Solid Propellants Area and/or the NASA/Douglas Spin Facility prior to beginning any operation. Only authorized personnel will be permitted to handle the rocket motor and/or pyrogen, i.e., GSFC or Thiokol personnel or personnel under their supervision. The number of personnel present will be held to the minimum required to perform the required operation.

2 Removal of components from the shipping container, inspection, and assembly or disassembly will be accomplished only in a designated facility in the PAA Solid Propellants Area or in the NASA/Douglas Spin Facility. The retromotor will be transported only in its shipping container.

3 The retromotor shipping container will be grounded at all times during storage. The retromotors are grounded to their container. During transfer of the motor from the container, it will always be electrically grounded. When mounted to the AIMP spacecraft, the retromotor, the retro thermal blanket, the spacecraft, and the pyrogen igniters are all electrically tied to each other and to ground.

4 Personnel handling the motor and/or igniter will be grounded by legstats to the same system that the retromotor and/or igniter is grounded to eliminate the possibility of static electricity buildup and discharge.

5 Personnel working on the motor or igniter will wear nonstatic producing, flame retardant coveralls or clean room smocks (the clean room smocks are nonstatic producing but are not flame retardant), legstats, and static free gloves.

6 Work areas will be covered with a conductive mat with ground straps attached to the building ground system.

7 Electrical resistance or continuity checks on the pyrogen igniters will be made with an approved Alinco Igniter Tester in the PAA igniter test area. Personnel will be protected from the pyrogen igniter by a steel and concrete barrier during this test.

8 The live pyrogen igniters will not be installed in the retromotor until F-1 day, just prior to fairing installation on Gantry 17A. The igniter shorting plug will only be removed when making the final connection of the igniter to the spacecraft harness. The spacecraft harness shorts the igniters when this connection is made.

9 No electrical tests will be performed on the igniter in or near the motor assembly.

10 The retromotor propellant autoignition temperatures are 1 hour at 400°F or 8 hours at 300°F. The pyrogen temperature limits are -60° to +190°F. The recommended continuous storage temperature is 80° $\pm$ 5°F. All sources of excessive heat and shock and sources of flame or spark must be prohibited from the retromotor and pyrogen igniter assembly.

11 The utmost care will be exercised in handling the retromotor and igniter to prevent mechanical damage and undue shock. The retromotor will be lifted and moved only with a crane, hydroset, and special hoisting adapter and only under the direct supervision of GSFC or Thiokol personnel.

12 The motor may be rested in its special handling scallop or a heavily padded surface but will not be rested on any flat hard surface which might scratch, dent, or overstress the case in a localized area.

13 There will be no metal working on the retromotor or pyrogen igniter or in the immediate vicinity.

14 The normal ordnance "common sense" practices will be enforced, i.e., (a) no matches, lights, or other flame or high temperature producing devices within 50 feet of the motor or igniter (b) only approved spark-proof flashlights utilized within 50 feet (c) no smoking within 50 feet (d) no electric tools or electric machinery used on or near the retromotor.

15 Fire fighting equipment and emergency medical treatment will be available in retromotor assembly and storage areas.

16 No static producing material will be utilized around the retromotor. Conductive (approved types) plastic will be utilized for clean room situations were necessary.

C. RETROMOTOR TRANSPORTATION.

Two TE-M-458 retromotors will be shipped by air cargo for the AIMP launch from Thiokol Chemical Company, Elkton, Maryland, to Cape Kennedy Air Force Station in accordance with established ICC regulations. The motors and igniters will be shipped in appropriately marked steel containers to:

PAA Supervisor of Solid Propellants

Cape Kennedy Air Force Station, Florida

TE-M-458 Retromotors

GSFC AIMP Program

ATTN: D. C. Sheppard/E. W. Travis

Building AE, Phone 853-2524

On arrival at Cape Kennedy Air Force Station, the motors and igniters will be placed in solid propellant storage and will remain in storage (except during X-ray de-perming, and igniter checks) until the initiation of preparations required prior to installation on the flight spacecraft in the NASA/DAC Area 39 Assembly Facility.

D. RADIOACTIVE SOURCE HANDLING

1. License. All sources are under GSFC Byproduct Material License No. 19-5748-2. The GSFC Radiological Officer will meet the spacecraft and associated radioactive sources when they arrive at ETR. The AEC approval for launch of the inflight radioactive sources will be forwarded upon its recipt.

2. Handling and Storage. Handling, storage, and use of the radioactive sources will be directly controlled by the GSFC Radiological Office. The GSFC Radiological Officer, or his representative, should be a member of the Impact Convoy.

The preflight checkout sources will be stored in Building AE when not being used in the Building AE work area, the Solar Simulation Facility (prototype spacecraft only), or on the spacecraft work levels of Complex 17A. The inflight sources are integral to the experiments aboard each spacecraft.

The spacecraft, all containers, and all work areas in Building AE, the Solar Simulation Facility, and Complex 17 will be monitored with appropriate survey meters and proper radiation signs will be posted. All personnel working with the spacecraft or the preflight checkout sources will be required to wear film badges and their exposure records will be maintained by the GSFC Radiological Office. Only authorized personnel will be permitted in the spacecraft area when the preflight checkout sources are in use in Building AE, the Solar Simulation Facility, or at Complex 17. The GSFC Radiological Office shall ensure that qualified personnel are continuously available while the spacecraft and the preflight checkout sources are located at ETR.

In the event of an incident which results in area contamination, the GSFC Radiological Office personnel will immediately notify the PAA Environmental Health Office, and will perform an immediate survey of the area to determine the extend and magnitude of the radioactive release. Access to such an area will be restricted to personnel required to prevent further damage to the spacecraft and associated equipment, or areas.

In the event of a land impact, a reasonable effort will be made to determine the condition of the radioactive material aboard the spacecraft and to effect its recovery. No attempt at recovery will be made in the event of a water impact.

3. Shielding. Type, manufacturer, model, and serial number of shielding to be used is not available at this time and will be nonexistant in several cases. The radioactive sources will be shielded during shipment and storage so as to comply with the requirements of Title 10, CFR, Part 20 and Title 49, CFR, Part 78.

E. DECONTAMINATION AND CLEAN ROOM ASSEMBLY

The AIMP spacecraft will be assembled under conditions that will minimize the possibility of contaminating the lunar surface with earth-origin microorganisms insofar as this can be done without jeopardizing spacecraft or mission reliability. A clean room will be provided at both GSFC and Cape Kennedy for component and spacecraft decontamination and clean assembly. The decontaminated spacecraft will be shipped in a sterile container to ETR.

Nonstatic producing clean room smocks that will be worn when working in close proximity to the spacecraft were procured from a KSC Safety Office approved vendor. These garments are not flame retardent because the treatment necessary to achieve flame retardant characteristics can be detrimental to maintaining a clean room environment.

F. ETR OPERATIONS SCHEDULE

This operations schedule describes the major spacecraft events that will occur during checkout and launch preparations. Prototype spacecraft events will be completed on F-9 day. The time designation as to when these operations will take place is presented in F minus days from launch. The following are the ground rules on which this schedule is based.

1 Only one spacecraft can be checked out at a time. Only one computer will be available to reduce the data.

2 Final spin balance must be accomplished by F-10 in order to give BTL adequate time to use the results in their calculations and make the necessary cross checks.

3 F-4 to F-0 are in accordance with launch vehicle operation.

4 Once the retromotor is installed, access to the payload will be limited.

5 Each experimenter is given a specific day to check his unit. On a non-interference basis he may check his experiment on days assigned to other experimenters.

6 It should also be remembered that once the retromotor is attached, it will require two days to remove and replace any experiment.

Day	Task
F-35	Decontamination team and project staff arrive. Delivery of spacecraft support trailers, first set of spacecraft GSE, and portable clean room.
F-34 to F-32	Preparation and cleaning of GSE for movement into clean room. Assembly of portable clean room. Biological monitoring started.
F-31	Prototype spacecraft and radioactive sources arrive at Building AE. GSE checkout.
F-30	Prototype instrument checkout. Biological samples taken of prototype spacecraft.
F-29	California prototype experiment checkout.
F-28	Iowa prototype experiment checkout.
F-27	MIT prototype experiment checkout.
F-26	GSFC thermal ion prototype experiment checkout.

Day	Prototype Spacecraft	Flight Spacecraft
F-25	Ames prototype experiment checkout.	Flight spacecraft and second set of GSE arrive.
F-24	GSFC magnetometer prototype experiment checkout.	Flight spacecraft biological samples taken. Spacecraft inspected.
F-23	Mechanical integration. Solar array paddle check insolar array building (may be interchanged with other tests to obtain suitable day).	Instrument checkout.
F-22	1. Mechanical integration and preparation for spin balance. 2. Magnetic check of spacecraft made and depermed if necessary.	MIT experiment checkout.
F-21		Iowa experiment checkout.
F-20		GSFC thermal ion experiment checkout.
F-19	1. Move to spin balance facility. 2. Decontaminate spacecraft/third stage interface. Take samples. 3. Mount to live third stage, put on dummy paddles. 4. Locate standoffs and tie-down brackets. 5. Remove from live third stage. 6. Move second set of GSE to gantry.	Ames experiment checkout.
F-18	1. Move to Building AE. 2. Decontaminate spacecraft retromotor interfaces. Take sample. 3. Mount dummy retromotor. 4. Decontaminate spacecraft/third stage interface. Take sample. 5. Mount on dummy third stage. 6. Check out second set of GSE and blockhouse wiring.	GSFC magnetometer experiment checkout.

Day	Prototype Spacecraft	Flight Spacecraft
F-17	1. Move spacecraft-dummy third stage combination to gantry and mount on vehicle. 2. Take biological samples.	California experiment checkout.
F-16	1. Check out spacecraft and blockhouse interface. 2. Check out spacecraft with GSE using F-1 day procedures. 3. Spacecraft data recorded. 4. Fairing wiped down and biological sample taken.	1. Spacecraft checkout. 2. Buy off flight experiments by experimenters.
F-15	1. Preliminary radio frequency interference (RFI) checks. 2. Spacecraft blockhouse control. 3. Install blockhouse and check out wiring. 4. Precautions taken to protect spacecraft biological integrity as fairing is installed. 5. Fairing installed and biological sample taken.	1. Mechanical integration. 2. Preparation for move to spin balance facility. 4. Retromotor magnetically checked and depermed if necessary.
F-14	1. Integration checks. 2. Biological sample taken.	1. Magnetically check spacecraft and deperm if necessary. 2. Move motor adapter ring to bunker area and check igniter wiring with live igniters. 3. Move to spin balance.
F-13	1. Spacecraft checks. 2. Biological sample taken.	1. Decontamination spacecraft/retromotor interface. Biological sample taken. 2. Mount and align live retromotor. 3. Bond on flight thermistors on retromotor. 4. Attach thermal blanket (do not tape).

Day	Prototype Spacecraft	Flight Spacecraft
F-12	1. Spacecraft checks. 2. Biological samples taken.	1. Attach dummy igniters. 2. Decontaminate spacecraft/third stage interface and take biological samples. 3. Mount spacecraft on third stage. 4. Check out despin function interface.
F-11	1. Spacecraft checkout. 2. Biological samples taken.	1. Mount flight paddles and booms. 2. Alignment and rough balance.
F-10	1. Spacecraft checkout. 2. Fairing installed. 3. Biological sample taken.	1. Flight balance. 2. Biological sample taken. 3. Paddles and booms removed.
F-9	1. RFI tests made. 2. Gantry removed and RFI tests made. 3. Fairing removed. 4. Prototype returned to clean room.	1. Standby. 2. Clean carrier.
F-8	1. Standby.	1. Mount dry nitrogen purge on carrier. 2. Move to gantry. 3. Mount on vehicle.
F-7	1. Standby.	1. Spacecraft checkout. 2. Biological sampling 3. Blockhouse checkout.
	Tracking and Data Systems Directorate Real Time Program Checkout	
F-6 F-5	1. Standby.	1. Blockhouse checks. 2. Spacecraft checkout. 3. Biological samples taken.

Day	Prototype Spacecraft	Flight Spacecraft
F-4	1. Standby.	1. Checkout of spacecraft and compatability check with vehicle systems. 2. Biological monitoring.
F-3	1. Standby.	1. Checkout of spacecraft. 2. Biological monitoring.

F-2 Day (Spacecraft Events)

Task	Time Required
Spacecraft system test	280 min.
Spacecraft stray voltage check	20 min.
DAC ordnance installation	*
Third stage squib installation	*
Spacecraft retromotor igniter installation and leak test	105 min.

Between F-2 and F-1 day the following spacecraft tasks are to be accomplished:

Remove spacecraft stripcoat and clean	420 min.
Spacecraft electrical check	60 min.
Paddle assembly and installation	120 min.
Sun gun check	20 min.

F-1 Day (Spacecraft Events)

Task	Time Required
FW-4 igniter hookup	*
Retromotor igniter hookup and thermal blanket installation	80 min.
Mechanical inspection and asepsis bag installation	30 min.
Fairing installation (Sometime during fairing installation remove asepsis bag.)	*

*To be determined by DAC and will be included in the Delta countdown.

F-0 day (Spacecraft Events)

1st task during DAC ordnance hookup will be spacecraft final arm and fairing access door sealed.	30 min.
At conclusion of DAC ordnance hookup, spacecraft will be exercised at discretion of spacecraft controller.	180 min.

SECTION V
GROUND SUPPORT EQUIPMENT

Information in this section will be supplied later as an addendum to this document.

APPENDIX A

AIMP RETROMOTOR
IGNITER INSTALLATION AND LEAK TEST
MARCH 1966

IGNITER INSTALLATION AND LEAK TEST

General

This procedure describes the installation of igniters into the retromotor for flight and the method for conducting a leak test to verify the pressure integrity of the motor assembly with igniters installed.

Personnel required:

GSFC Mechanical Engineers (2)
GSFC Mechanical Technician

Equipment required:

Two igniter assemblies, TCC #E17466-01
Igniter O-rings, MS 28775-012
O-ring lubricant, Dow Corning #55M
Deep well socket wrench
Torque wrench (60 in-lb)
Safety wire - .032" diameter
Safety wire pliers
Leak test suitcase
Acetone
Kimwipes
Cotton swabs
Watch
1 Container of liquid nitrogen (LN_2) with a 4-foot fill hose

Operational Procedure

_____1. Secure permission from the area supervisor to remove the two igniter assemblies from the storage area.

_____2. Open the cap on the top of the shipping container and remove the ground strap from each igniter.

_____3. Transport the igniter assemblies in their cannisters to the gantry via an approved vehicle.

_____4. Carry the ingniters and equipment required to level 9 of the gantry.

____ 5. Position the leak test suitcase approximately 8 feet from the vehicle centerline. Check to insure that the suitcase is close enough to allow the plug to be placed into the motor nozzle.

____ 6. Close valves A, B, and C on the leak check gear. See figure A-1. Disconnect the hose. (Leave hose disconnected until step 20.)

____ 7. Fill both LN_2 containers in the leak check suitcase.

____ 8. Replace the cover on the leak check suitcase.

____ 9. Verify that the automatic relief valve on the suitcase is working.

____ 10. Unpack the igniters; record igniter serial numbers on this page and in the MSB Launch Procedures and Checkout Book.

	S/N	Location
Igniter A	____	________
Igniter B	____	________

NOTE: DO NOT REMOVE IGNITER SHORTING PLUGS.

____ 11. Secure O-rings for each igniter from the envelopes inside the igniter cannister and inspect for damage.

____ 12. Remove dummy igniters from the retromotor.

____ 13. Inspect the igniter ports in the aft closure and clean the ports using cotton swabs and alcohol, if necessary.

____ 14. Lightly grease the O-rings with Dow Corning #55 lubricant.

____ 15. Slip the O-rings in place on the igniters.

____ 16. Install both igniters finger tight, locating the igniters as recorded in step #10.

____ 17. Torque each igniter to 60 in-lb using the deep well socket and torque wrench.

____ 18. Lockwire each igniter squib to the aft closure lockwiring. Recheck igniter torque.

____ 19. Lightly grease the O-ring on the leak test motor plug using Dow Corning #55 lubricant.

_____ 20. Remove suitcase cover and attach hose to vacsorb pumps with the quick disconnect fitting.

_____ 21. Open valve C, near the gage.

_____ 22. With a finger resting lightly over the hole in the motor plug, open valve A slightly to verify suction. Close valve A immediately after verification.

_____ 23. Remove the aft closure rat plug from the nozzle.

_____ 24. Insert the motor plug into the nozzle.

_____ 25. Open valve A slightly until the vacuum reaches 27 #1 inches of mercury, then close the valve. Record pressure. Close valve C.

P= _______________ inches of mercury

_____ 26. Disconnect hose from the pump at the quick disconnect.

_____ 27. Close the cover on the suitcase and check automatic valve operation.

_____ 28. Should Vacsorb A be inoperative or incapable of pumping to the required level, close valve A and use Vacsorb B by opening valve B.

_____ 29. Wait 5 minutes for the system to stabilize. Vacuum must be 27 ± 1 inches of mercury. Record the pressure.

P= _________________ inches of mercury

_____ 30. Wait for 10 minutes. The pressure shall not increase more than 1/2 inch of mercury from the pressure recorded in step #29. Record the pressure.

P= __________ inches of mercury.

_____ 31. Holding onto the pipe attached to the motor plug, open valve C to vent the motor.

_____ 32. When the motor is vented, remove the motor plug and store it in the suitcase.

_____ 33. Inspect motor rat plug for damage and wipe the top and bottom aluminum foil with acetone.

_____34. Press the rat plug into the nozzle until it seats fully.

_____35. Notify the blockhouse that the igniter installation and leak check are complete.

_____36. Remove leak check gear and other equipment from the gantry.

CAUTION: BE CAREFUL NOT TO SPILL THE LN_2 IN THE LEAK CHECK SUITCASE.

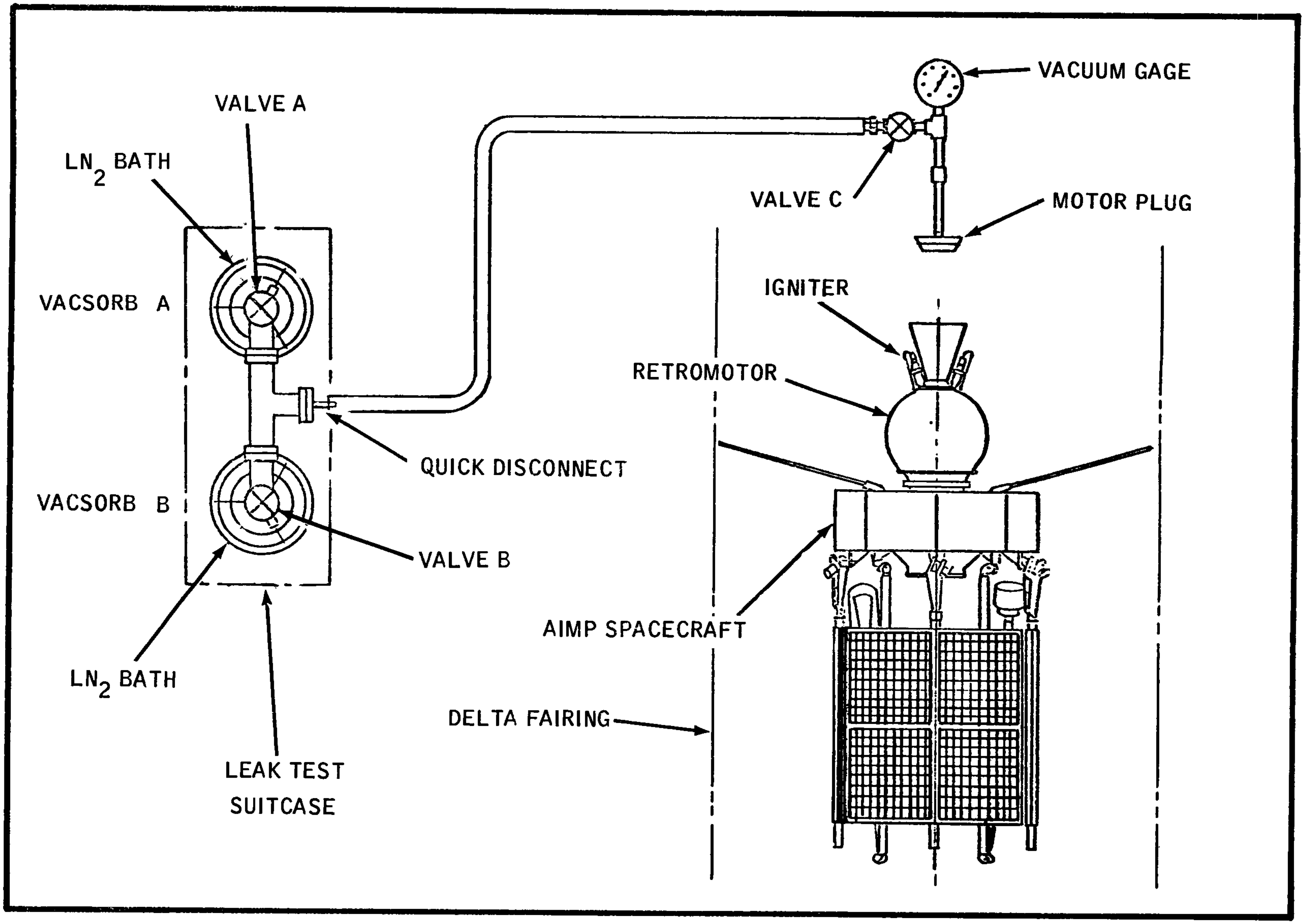


Figure A-1. Retromotor Gantry Leak Check Setup

APPENDIX B

AIMP RETROMOTOR X-RAY PROCEDURES

APPENDIX B

X-RAY PROCEDURES

General

The purpose of this operation is to x-ray the motor, less igniters, to insure that there are no separations or cracks in the grain configuration and that any discrepancies in the motor case will be detected.

The x-rays are taken in the Solid Propellants area and consist of nine views to duplicate x-rays taken prior to shipment from the Thiokol Elkton Division. Motor handling for these checks is under the supervision of a GSFC Mechanical Engineer.

Personnel required:

Solid Propellant Area
GSFC Mechanical Area
GSFC Mechanical Technicians

Equipment required:

TE-M-458 retromotor assembly
Retromotor handling dolly and lifting rig
Conductive mat attached to building ground system
Conductive shoe covers for personnel
Conductive strap to be connected between motor aft closure bolt and building ground system
X-ray equipment as required for the views noted in figures B-2 and B-3.
Two wooden x-ray pallets

Operational Procedure

____1. Secure permission from the area Supervisor to remove the motor from storage and move it to the designated facility for x-ray checks.

____2. Move the motor shipping container into the designated area as close to the work area as practicable.

____3. Place a conductive mat in the work area and connect the mat to the building ground system.

____4. Personnel to handle the motor must wear legstats.

_____5. Remove the shipping container cover by lifting the cover straight up.

_____6. The retromotor and igniters are grounded to the shipping container as shown in figure B-1. The motor is covered with a velostat protective bag.

_____7. Attach a conductive strap between an aft closure bolt and the building ground system. Disconnect the shipping ground straps at points A and B (figure B-1).

_____8. Locate the motor handling dolly or plywood holding board within the field of view of the X-ray device.

_____9. Lift the motor from the shipping container and set it in an upright position on the wooden pallet. The motor may be lifted by hand or by crane using the lifting cradle stored in the motor container.

_____10. Remove the motor lifting cradle and pull the velostat bag down flush with the aft mounting ring.

_____11. Reassemble the lifting cradle to the motor, lift the motor off the pallet by hand or by crane, and pull the velostat bag from under the motor.

_____12. Set the motor back on the wooden pallet and remove the lifting cradle.

_____13. X-ray the retromotor, taking the views specified in figures B-2 and B-3. The machine settings, film data, etc. included in the block on figures B-2 and B-3 are the settings used for the machine at Thiokol - Elkton and are included for reference only.

_____14. Criteria for flight acceptance of a retromotor is contained in Thiokol specification number P 20025

_____15. Attach the lifting cradle to the motor.

_____16. Lay the velostat bag across the second wooden motor pallet and set the motor onto the velostat.

_____17. Remove lifting cradle and pull velostat bag up over the motor.

_____18. Reassemble the lifting cradle around the motor.

_____19. Transport the motor to the shipping container and set the lifting cradle with the retromotor into the shipping container.

_____20. Attach the shipping container ground straps to points A and B (figure B-1).

_____21. Remove the ground strap running from the aft closure bolt to building ground.

_____22. Close and secure the container.

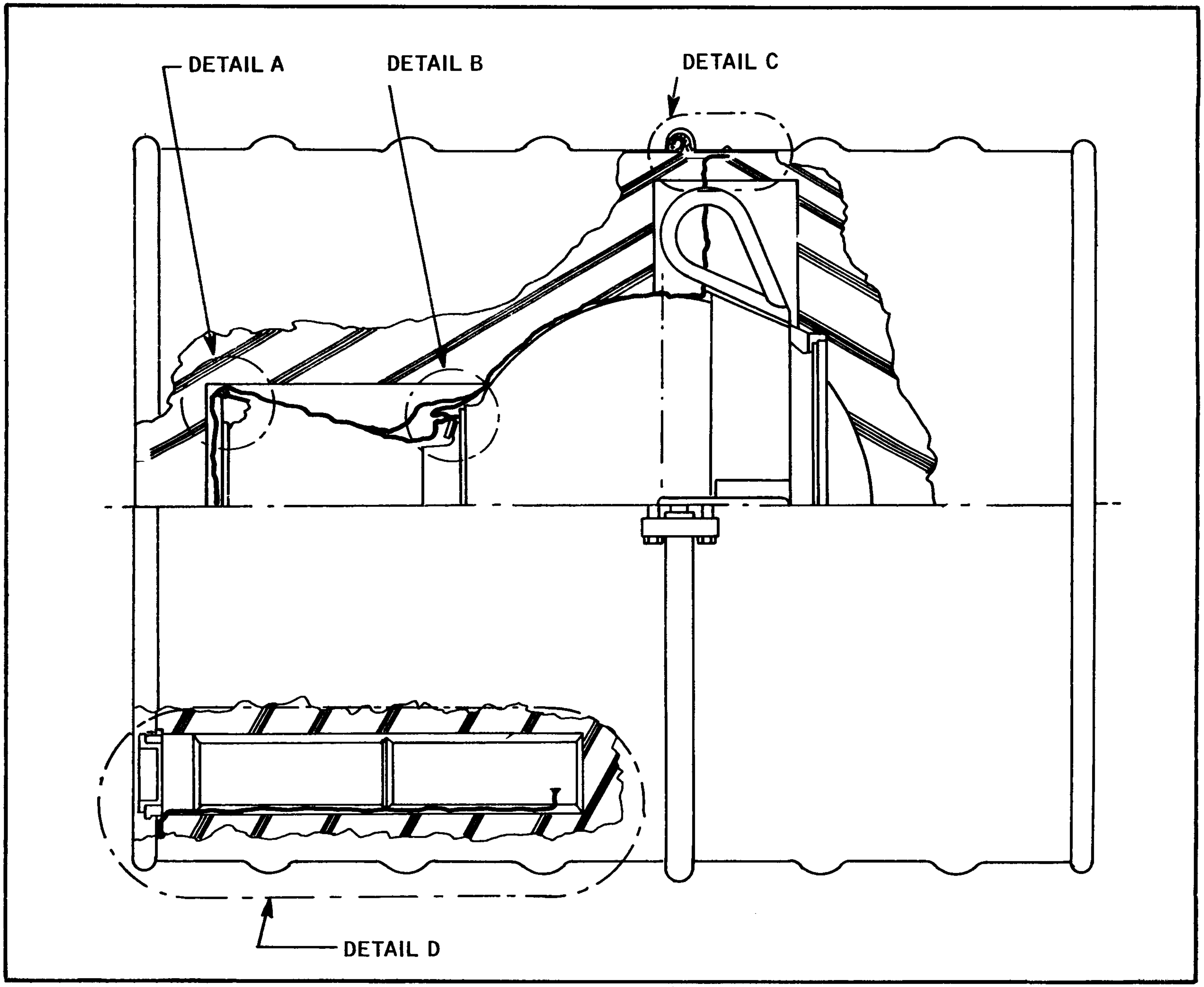


Figure B-1. Retromotor Grounding in Shipping Container (Sheet 1 of 2)

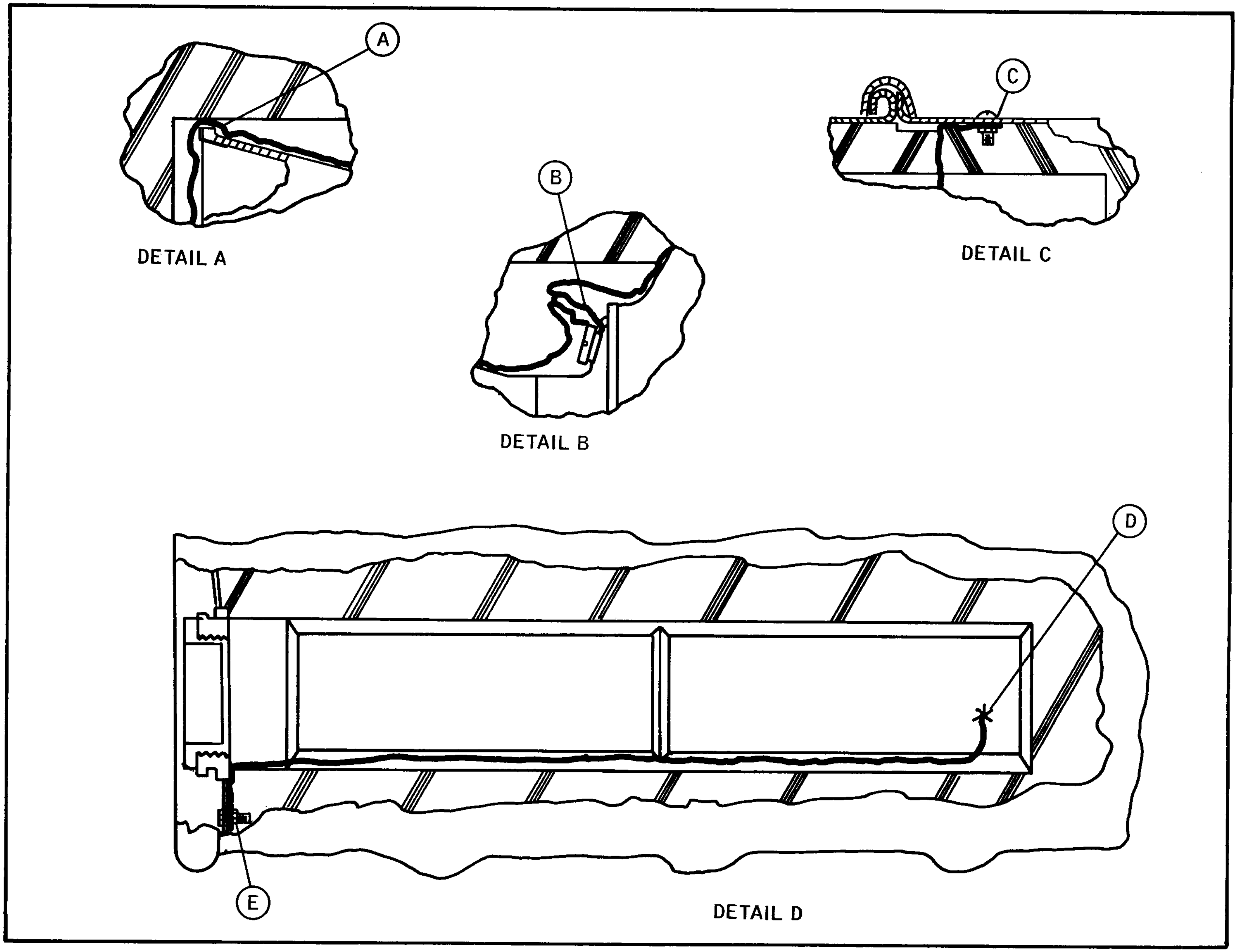


Figure B-1. Retromotor Grounding in Shipping Container (Sheet 2 of 2)

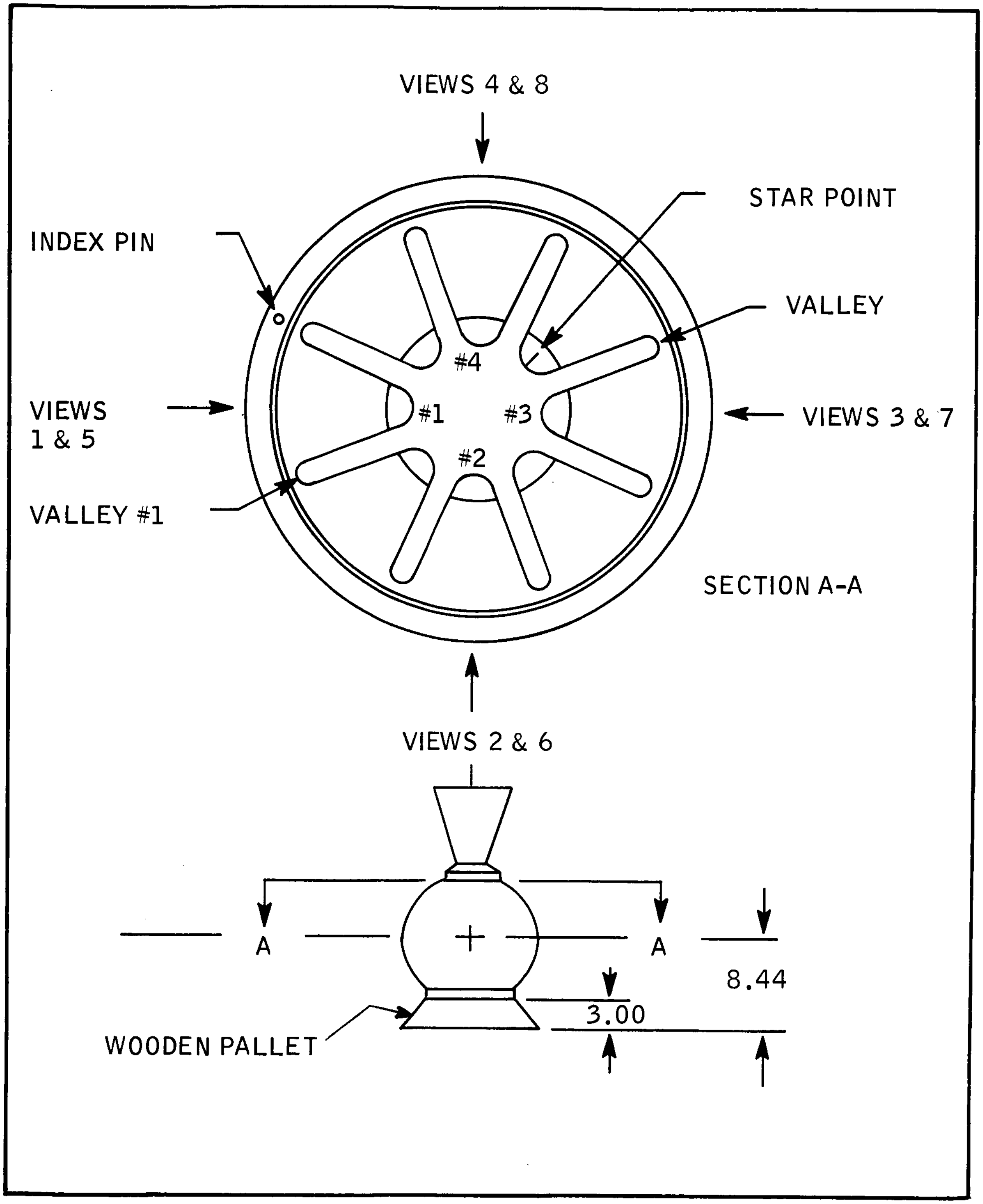


Figure B-2. Retromotor in Vertical X-ray Position

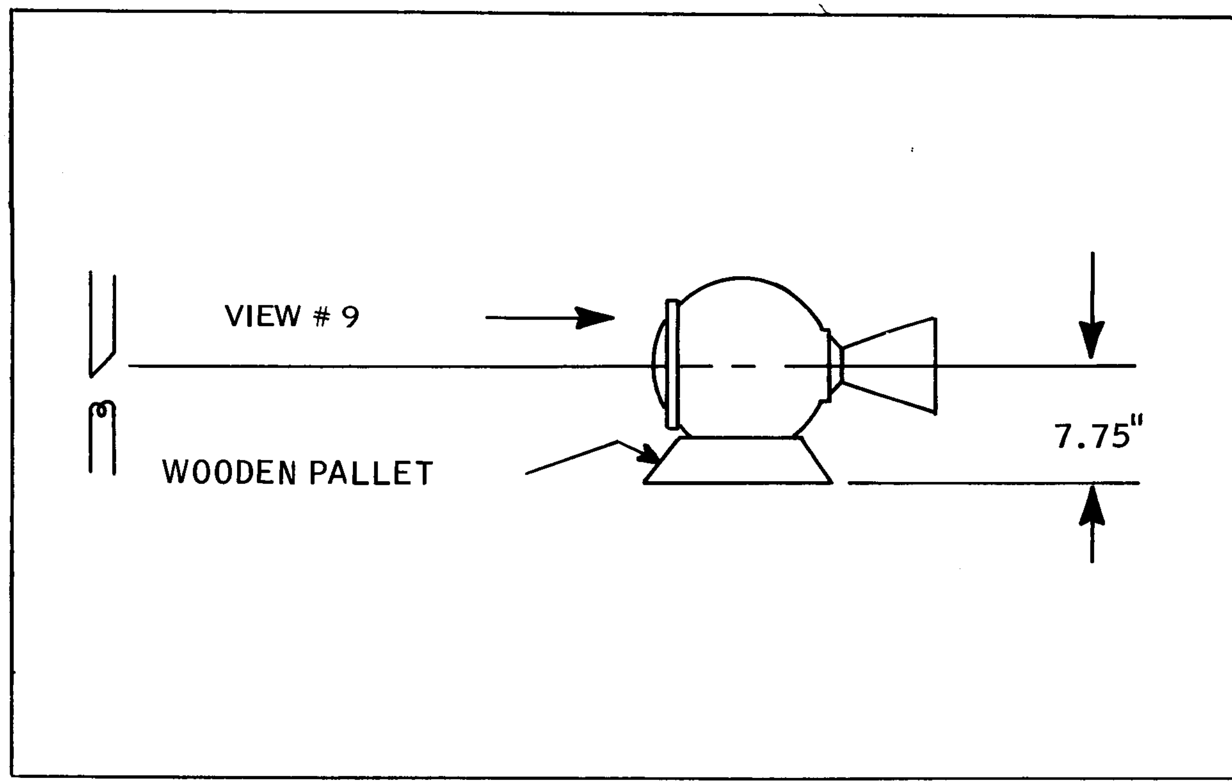


Figure B-3. Retromotor in Horizontal X-ray Position

APPENDIX C

AIMP RETROMOTOR MAGNETIC DEPERMING

APPENDIX C

RETROMOTOR MAGNETIC DEPERMING

General

The procedure outlines the method for magnetically deperming the AIMP retromotor assembly with live igniters mechanically installed but electrically shorted.

Personnel required:

Solid Propellants Area

GSFC Mechanical Engineer
GSFC Mechanical Technicians (2)
GSFC Magnetic Properties Engineer
GSFC Magnetic Properties Technician

Equipment required:

AIMP retromotor
Two igniter assemblies
Wooden pallet
Retromotor lifting sling
Retromotor deperm adapter
GSFC deperm coil and dolly
Variac, type WSOHM
DC power supply, harrison labs #808A, p5
Retromotor mounting bolts - 8 #1/4 - 28 x 7/16 long
Third stage clamp band - aluminum
Ground strap, non-magnetic
Conductive mat connected to building ground system
Conductive legstats for personnel
Forster/Hoover magnetometer, #MF-55-331-10X
Safety wire - .032 diameter, aluminum
Safety wire pliers

Operational Procedure

____1. Secure permission from area supervisor to conduct magnetic deperming on motor in a designated area within the solid propellants facility.

____2. Perform a functional checkout on the sensor and deperming equipment.

____3. Move the motor in its shipping container into the designated area. Locate the motor shipping container as close to the work area as practicable.

____4. Locate a conductive mat connected to the building ground system in the motor work area. Connect a non-magnetic conductive ground strap from the deperm dolly to the building ground system.

____5. Personnel to work on the motor must wear legstats.

____6. Clear the area of all unnecessary personnel.

____7. Remove the cover of the motor shipping container by lifting the cover straight up.

____8. Connect a non-magnetic conductive strap, tied to the building ground system, to the motor case on an aft closure bolt.

____9. Disconnect the shipping ground straps at points A and B. See figure C-1.

____10. Lift the motor by hand from the shipping container using the lifting cradle stored in the shipping container with the motor.

____11. Set the motor on the wooden pallet.

____12. Remove the lifting cradle.

____13. Peel the velostat bag from around the motor down to the mounting ring.

____14. Reassemble the lifting cradle to the motor.

____15. Assemble the deperm adapter to the deperm dolly.

____16. Place motor mounting bolts (eight #1/4-28 x 7/16 hex head) on the deperm dolly.

____17. Lift the motor from the wooden pallet using the cradle sling and a crane.

____18. Lower the retromotor onto the deperm dolly.

____19. Mount the motor to the dolly using the eight bolts provided.

____20. Remove the motor lifting cradle and sling.

____21. Perform magnetic measurements and deperming as required. Record the magnetic measurement motor s/n __________ Magnetic__________GAMMA at __________inches.

____22. Attach motor lifting cradle to the motor and the sling to a crane.

____23. Remove the eight mounting bolts.

____24. Lift the motor from the dolly using the crane.

____25. Place the velostat bag across the wooden pallet.

____26. Lower the motor onto the pallet.

____27. Remove the lifting cradle and sling.

____28. Pull velostat bag up over the motor.

____29. Reassemble the lifting cradle to the motor.

____30. Lift the motor by hand and set it into the shipping container.

____31. Connect the shipping container ground straps at points A and B. See figure 1.

____32. Disconnect the ground strap connected to the building ground.

____33. Pull velostat bag up over the nozzle.

____34. Put the shipping container top back in place and bolt closed.

____35. Remove both igniter containers from their storage cavity in the shipping container.

____36. Connect a non-magnetic conductive strap, tied to the building ground system, to each igniter.

____37. Disconnect the shipping container ground strap from the igniters.

____38. Remove the red warning tag from the shorting connector by cutting the safety wire.

NOTE: DO NOT REMOVE THE IGNITER SHORTING CONNECTORS

____39. Reassemble the warning tag to the shorting connector using aluminum safety wire.

____40. Magnetically map and deperm the igniter assemblies as required.

____41. Record the magnetic properties of the igniters:
s/n_____, _____GAMMA @_____inches
s/n_____, _____GAMMA @_____inches

____42. Reconnect the shipping container ground strap

____43. Disconnect the ground strap to building ground.

____44. Put the igniters in their containers.

____45. Place the containers in their cavity in the motor shipping container.

____46. Lockwire the cap on the shipping container.

____47. Repeat the process for the flight back-up motor using another copy of this document.

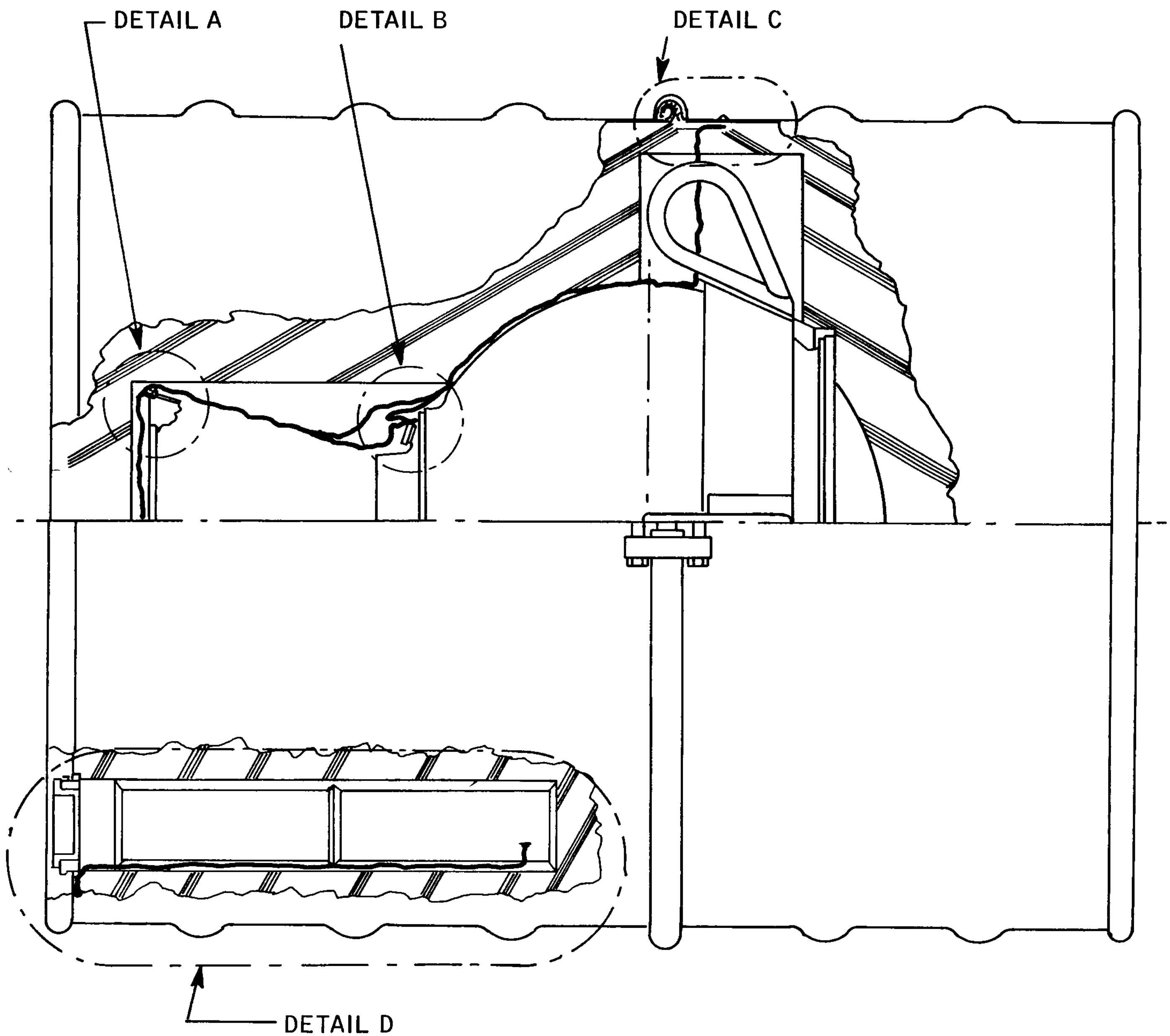


Figure C-1. Retromotor Grounding in Shipping Container (Sheet 1 of 2)

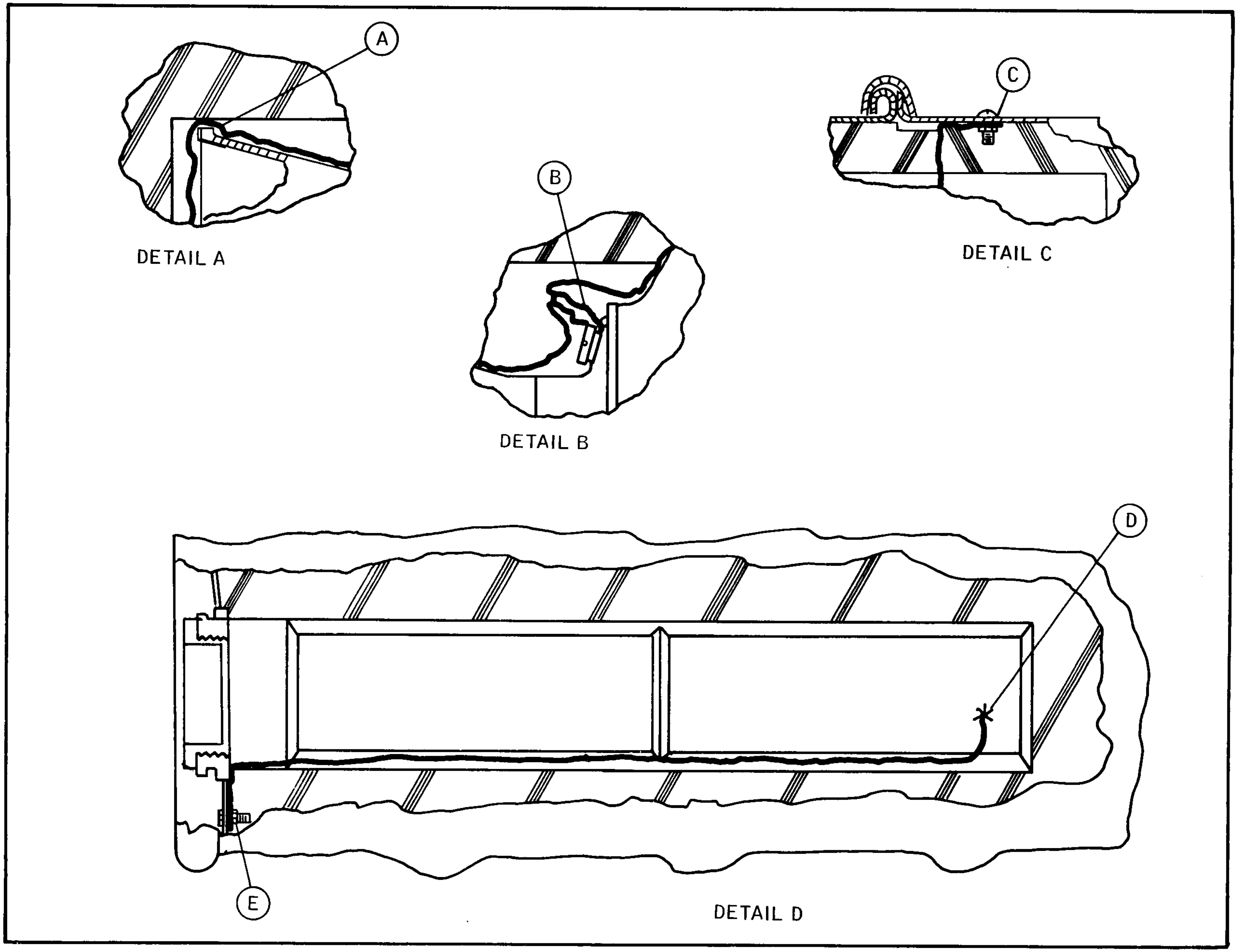


Figure C-1. Retromotor Grounding in Shipping Container (Sheet 2 of 2)

APPENDIX D

INSTRUCTION MANUAL
FOR
FORSTER/HOOVER ELECTRONICS, INC.
MAGNETOMETER
TYPE MF-55-331-10X
MANUFACTURED FOR
NASA, GODDARD
CONTRACT NAS5-2761

1. GENERAL DESCRIPTION

The Forster/Hoover Electronics Magnetometer Type MP-55-331 consists of three transistorized, standard 19-inch rack and panel mounted, modular constructed, precision magnetometers capable of measuring magnetic fields in the range of 0.1 millioersted to 1000 millioersteds with a resolution of 1×10^{-3} millioersted.

The instrument accuracy is $\pm 1\%$ of full scale on any range that is calibrated. Built-in calibration is accomplished on ranges 1, 10, 100, and 1000 millioersted with sensitivity switch at 1X or at 10X for full scale deflection. Intermediate ranges are calibrated whenever the next lower power of 10 range is calibrated and accuracy is $\pm 3\%$ full scale. Use of external calibration standards will improve intermediate ranges to $\pm 1\%$ accuracy.

Power for energizing the unit is from 110 volt, 60 cps current. Power consumption is approximately 12 watts, including the probe tripole.

The compensator is a highly stable dc power supply whose output is switched to give precise amounts of current for compensation. Because direct switching is used, the range covered by each switch position is dependent upon the current being drawn through the switch combination. As a result, the unit has been designed to permit an overlap in positions of the switches so that no hole exists. Because of the overlap permitted,the switch positions do not precisely give the magnitude of compensation field in mOe. They do indicate within a few percent the actual compensating field. Also, because of the stable power supply and the precision switches, the settings are repeatable with a high degree of accuracy.

The filter is a time integrating circuit that permits the lower frequencies to pass through to the output, and attenuates 60 cps magnetic fields and its harmonic. The filter is desirable when very low dc fields are to be measured in the presence of a relatively larger ac ambient. The long time constant chosen for this circuit permits measurements on the most sensitive ranges, even when the ambient earth field produces fluctuations and short time variations. If the time constant is too long, it can be reduced by changing the value of capacitor Cp, which should be of the non-polarized type.

An amplifier has been added to increase the sensitivity of all ranges by a factor of 10. This is introduced whenever the sensitivity switch is turned to X10. This permits measurements of ± 10 gamma full scale.

Recorder output requires an input impedance of 100k ohms or higher, so as not to load the source impedance of 10k ohms.

2. PRINCIPLE OF OPERATION

The magnetometer and probe design is based on the second harmonic principle of operation. The probes are the heart of the system and are special Forster probes, providing high sensitivity, accuracy, linearity, and stability in a markedly small size. The probes have two highly permeable cores around each of which is wound a primary and secondary winding. The primaries are connected series-opposing and the secondaries series-aiding. When a low frequency current is fed to the primaries and a zero magnetic field exists along the core axes, the voltage induced in the secondaries is self cancelling because of symmetry, one secondary having a voltage equal but opposite in phase to the other, producing a net output voltage of zero.

When a dc magnetic field acts parallel to the cores, an unbalance occurs in that the secondary voltages do not cancel exactly, producing a voltage which is (1) proportional in amplitude to the magnetic field acting parallel to the cores and (2) is of a frequency twice that of the primary. This twice fundamental frequency, or the second harmonic, voltage is filtered, amplified, phase rectified, and fed to the output.

3. DESCRIPTION OF MAGNETOMETER OPERATION

The block diagram of the magnetometer (figure D-1) shows the operation of the instrument. A 10 kc crystal oscillator (providing a stable source of frequency) is fed to a buffer amplifier, which amplifies the 10 kc voltage sufficiently to drive the probe primaries. If a magnetic field exists parallel to the probe axes, a 20 kc signal is produced whose amplitude is proportional to the magnetic field. The 20 kc signal is passed through a filter and a tuned amplifier, to the phase controlled rectifier. Simultaneously, a portion of the 10 kc voltage from the buffer amplifier is fed to a frequency doubler stage where the 10 kc becomes 20 kc for a reference voltage in the phase rectifier, and a dc voltage is produced which has a polarity, plus or minus, depending upon whether the 20 kc signal voltage is in phase or 180° out of phase with the reference voltage. The dc voltage amplitude is proportional to the magnitude of magnetic field acting parallel to the probe cores.

4. PANEL LAYOUT

The following items are located on the front panel:

1 Meter - Zero center meter with 100 units (50 scale divisions) either side of zero, mirror backed, 1% full scale linearity

2 Meter Adjustment - Screw drive adjustment for mechanical zero of meter needle

3 Calibration pushbutton - Depressing button allows calibration current to probe

4 Calibration adjustment - Screw driver adjustment for proper deflection as described in calibration procedure.

5 Output selector - M, meter only; MO, meter and recorder output simultaneously; 0, recorder output only

6 Recorder Output - front panel - ac and dc

7 Power switch - Controlling dc power supply input to magnetometer

8 Probe connector - Eight pin receptable to receive probe cable

9 Polarity switch, compensation - plus, minus, and off

10 Filter - 60 cps filter off and on switch

11 Sensitivity - X1 Normal; X10 increases sensitivity of all ranges by 10 times. Thus full scale ranges become 100, 50, 20, 10, 5, 2, 1, 0.5, 0.2, 0.1 mOe

12 Range switch - 11 position rotary switch with 1, 2, 5, 10, 20, 50, 100, 200, 500, and 100 mOe full scale ranges and 0 position (sensitivity off).

13 Compensation 100's - Each step corresponds to approx 100 mOe

14 Compensation 10's - Each step corresponds to approx 10 mOe

15 Compensation 1's - Each step corresponds to approx 1 mOe

16 Compensation 0-1 - ten turn pot permitting compensation 0 to 100 gamma, resolution 1 part in 1000.

On the back panel are located: (1) eight pin probe connections in each channel identical to eight on the front panel, (2) a 115 vac line plug connection, and (3) ac and dc recorder outputs for each channel. Dc output signals are obtained on the upper jack in each channel for recording. Source impedance is 10,000 ohms, thus recorders having input impedances of 100 k ohms or higher should be used. Ac fields modulating the 20 kc carrier signal are obtained on the lower output jack. This output can be connected to a scope so that ac modulation of the 20 kc carrier can be seen.

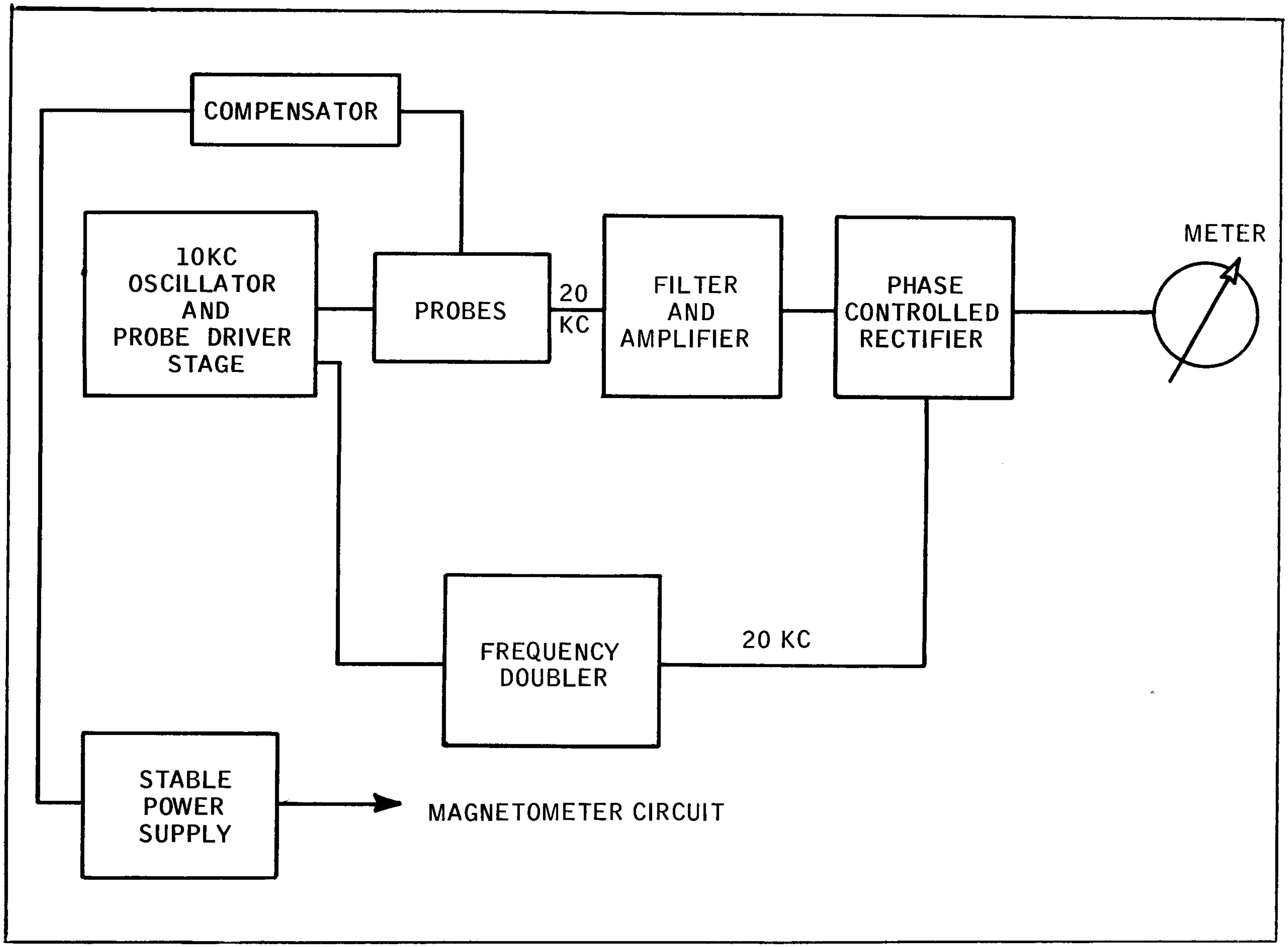


Figure D-1. Magnetometer Compensator Block Diagram

APPENDIX E

DEPERM COIL LOCATION
DEPERM COIL BLOCK DIAGRAM

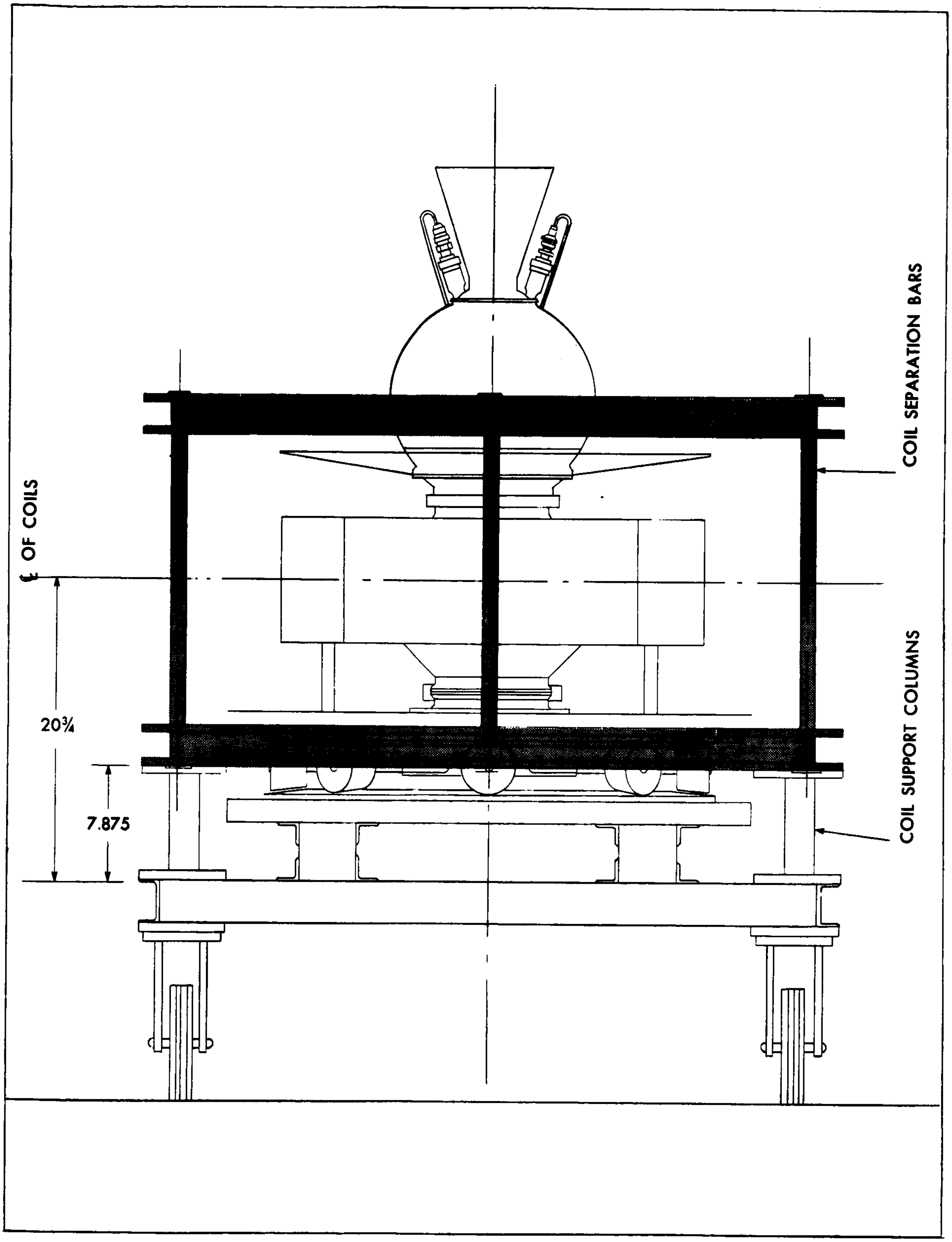


Figure E-1. Deperm Coil Location

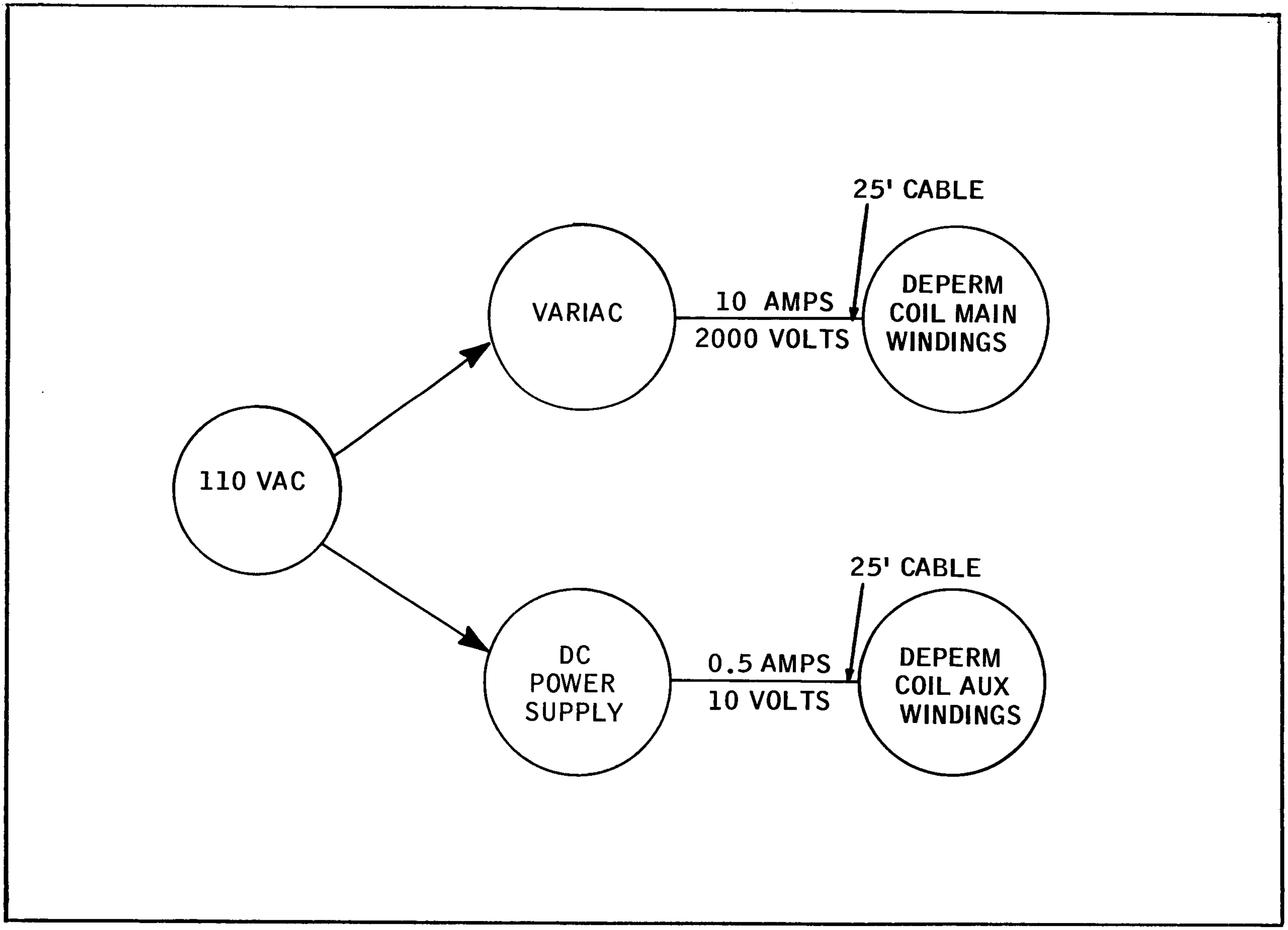


Figure E-2. Deperm Coil Block Diagram